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SOFT LANDER PART I

Mars Soft Lander Capsule Study (Entry From Orbit) - Summary

Prepared by

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for Langley Research Center

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Mars Soft Lander Capsule Study (Entry From Orbit) - Summary

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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1. SUMMARY

The Mars Soft Lander Capsule Study (Entry from Orbit) is one of several studies being conducted for the Langley Research Center to support planning of Mars exploration in the 1970's. Parametric studies have shown that a capsule weighing less than 1500 pounds can perform a significant scientific mission despite the present uncertainty of the Martian environment. A weight reduction of 15% and compatibility with an existing Titan payload shroud can be achieved if present severe environmental constraints are somewhat modified. Such a capsule would obtain atmospheric measurements during the entry and descent phases, acquire facsimile images and measure the composition of the surface after landing, and repeatedly measure the atmospheric temperature, pressure, and humidity and the wind velocity during one diurnal cycle. Engineering and scientific data would be relayed to earth via the orbiter. Surface mission duration can be extended to 90 days or longer for an 8 to 10% increase in weight.

Differences in capsule characteristics caused by varying lifetime or environmental constraints are exhibited by four conceptual designs which have been prepared to supplement the parametric studies. The estimated weight of these four ranges from 1200 pounds to 1600 pounds as shown in Figure 1-1. Concept I is the minimum weight capsule. It is designed for a one day mission and modified environment constraints. Concept II also accomplishes a one day mission, but utilizes the more conservative nominal environment and flexibility constraints. Concept III also uses the nominal constraints, but has extended lifetime capability. Extended lifetime together with modified constraints is provided by Concept V.

The soft lander study is an extension of earlier studies of Mars exploration capsules; the VOYAGER Capsule Phase B study provided both data and methodology for this study. Emphasis during the soft lander study was concentrated on those aspects particularly pertinent to the selection of a capsule concept for the 1973 launch opportunity. Cost trends, which are expected to play a significant part in this selection, have been established parametrically.

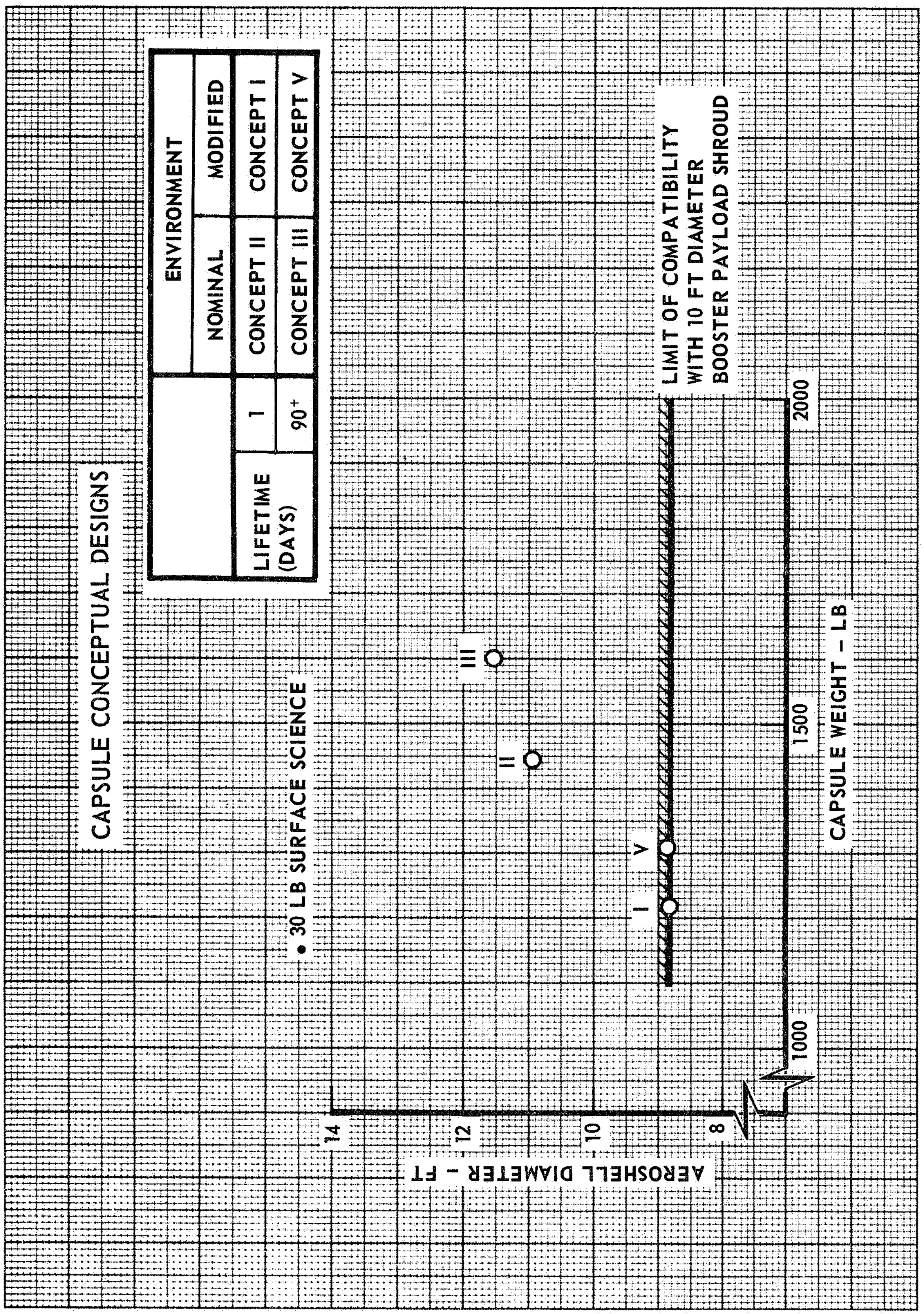

CAPSULE CONCEPTUAL DESIGNS
• 30 LB SURFACE SCIENCE
ENVIRONMENT
NOMINAL
MODIFIED
LIFETIME (DAYS)
1
CONCEPT II
CONCEPT I
90+
CONCEPT III
CONCEPT V
III
II
V
I
LIMIT OF COMPATIBILITY WITH 10 FT DIAMETER BOOSTER PAYLOAD SHROUD
AEROSHELL DIAMETER – FT
14
12
10
8
1000
1500
2000
CAPSULE WEIGHT – LB

FIGURE 1–1

The range of capsules to be studied was limited by a minimum required mission at one extreme and by a maximum capsule diameter at the other. The minimum mission consisted of entry and surface atmospheric measurements, post-landed imaging, soil composition measurements and transmission of ten million bits of data during a surface lifetime of at least one day. Maximum aeroshell diameter was limited to 178 inches to allow use of booster payload shrouds which are no greater than 16 feet in diameter. A 10 foot payload shroud would eliminate the need for hammerheading; this criterion was used to size Concepts I and V as shown in Figure 1-1.

The constraints which have guided most of the recent Mars capsule studies were used for the soft lander study; they are summarized in Table 1-1. The VOYAGER Environmental Predictions Document (SE003BB001-1B28), October 26, 1966, and the 1973 VOYAGER Capsule Constraints and Requirements Document (SE002BB002-2A21), Revision 2, June 12, 1967, were used to establish nominal environments; parametric evaluation of the effects of the environment definition on capsule design and operation was a part of the study.

TABLE 1–1

SPECIFIC DESIGN AND OPERATIONAL CONSTRAINTS AND ASSUMPTIONS

	INITIAL	MODIFICATIONS
DESIGN		
CAPSULE		
SHAPE:	SPHERE CONE	
WEIGHT:	1000 TO 5000 POUNDS	
AEROSHELL DIAMETER:	16 FT MAXIMUM	
STERILE CANISTER DYNAMIC ENVELOPE:		15 FT 4 IN. MAXIMUM
OPERATIONAL		
LAUNCH, INTERPLANETARY CRUISE, & MARS ORBIT		
OPPORTUNITIES:	1973, 1975, 1977	EMPHASIS ON 1973
TRAJECTORY TYPE:	TYPE I AND TYPE II	
LAUNCH WINDOW	30 DAY MINIMUM BASED ON C_3 (PREFERRED)	COMPATIBLE WITH LANDING AT $\phi = 10°N$, $i = 60°$, $\Gamma_s = 60°$
FIRING WINDOW:	1 HR MINIMUM DAILY (PREFERRED)	
DECLINATION, OUTGOING:	5° TO 36° (PREFERRED)	5° TO 50°
INCLINATION TO ECLIPTIC:	> 0.1 DEG	
HYPERBOLIC EXCESS SPEED AT MARS:	3 KM/SEC MAXIMUM (PREFERRED)	
MARS ORBIT SIZE:	SYNCHRONOUS, NOMINAL 25 HR PERIOD SEMI-SYNCHRONOUS, NOMINAL 12.5 HR	SYNCHRONOUS ONLY
MARS ORBIT INCLINATION		INCLUDE $i = 60°$
SEPARATION, DESCENT, AND LANDING		
ENTRY ALTITUDE:	800000 FT	
ENTRY VELOCITY:	13000 TO 15200 FPS (INERTIAL)	
ENTRY FLIGHT PATH ANGLE:	GRAZE TO −20 DEG	
ATMOSPHERES:	VM–1 THROUGH VM–10	
AERODYNAMIC DECELERATOR DEPLOYMENT:	PARACHUTE AT MACH 2. ATTACHED INFLATABLE DECELERATOR AT MACH 6	
SURFACE SLOPE:	UP TO 34° ABOVE LOCAL HORIZONTAL	
LANDING SITE		NEAR EQUATORIAL, 60° SOLAR ANGLE
POST-LANDING		
EXPERIMENTS:	TO BE DEFINED BY MCDONNELL & APPROVED BY NASA	
EARTH COMMUNICATIONS:	COMPATIBLE WITH DEEP SPACE NETWORK 10^7 BITS MIN; 10^8 BITS DESIRED	

1.1 Capsule Operation

The capsule mission starts with boost into Earth orbit; it continues with a several month transit to Mars and then insertion into a Mars orbit as shown in Figure 1.1-1. For the 1973 opportunity, the launch will occur in the mid-summer and arrival will occur in the early spring of 1974. Possible launch dates and transit times are shown in Figure 1.1-2.

After achieving the Mars orbit, the capsule separates from the orbiter, performs a propulsive deorbit maneuver, descends, and enters the atmosphere. Approximately 92% of the entry velocity is dissipated due to atmospheric drag on the aeroshell. As illustrated in Figure 1.1-3, a parachute is used to provide further deceleration and aeroshell separation prior to ignition of the terminal propulsion system; however, some candidate concepts include other auxiliary aerodynamic decelerators or do not require one. The terminal propulsion system decelerates the capsule at low altitude. The lander then free-falls to the surface, absorbing the landing shock loads in a crushable impact attenuator. UHF transmission of entry science and engineering data occurs throughout the descent, entry, and landing phases; storage and delayed transmission of data is used to eliminate loss of information due to entry communications "blackout".

After landing, the facsimile camera and atmospheric experiments are initiated immediately and data is transmitted to the orbiter. Careful selection of descent trajectory will provide several minutes of post-landing transmission capability. Operation of the soil composition instrument, an alpha spectrometer, will be initiated, but several hours are required to obtain useful data and this will prevent first day data transmission. The capsule is heated during the Martian night by isotopes or electrical heaters.

Relay communication on the following day(s) occurs when the orbiter is near periapse. For long lifetime landed missions, the availability of the orbiter for communications relay cannot be readily assured; S-band transmission direct to Earth and a command link will probably be required. Surface mission duration for soft landers is primarily a function of the type of power selected.

TYPICAL MISSION PROFILE

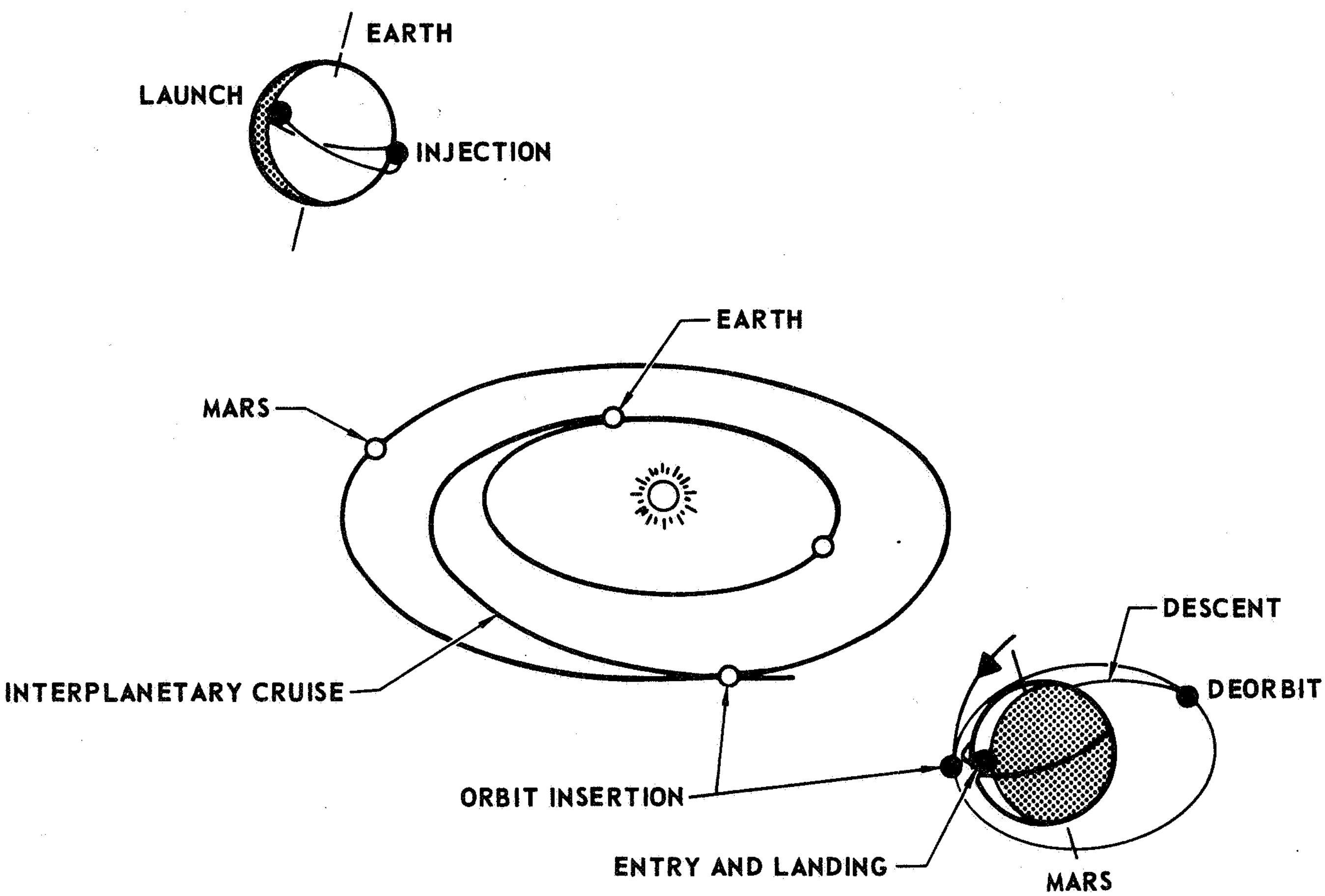
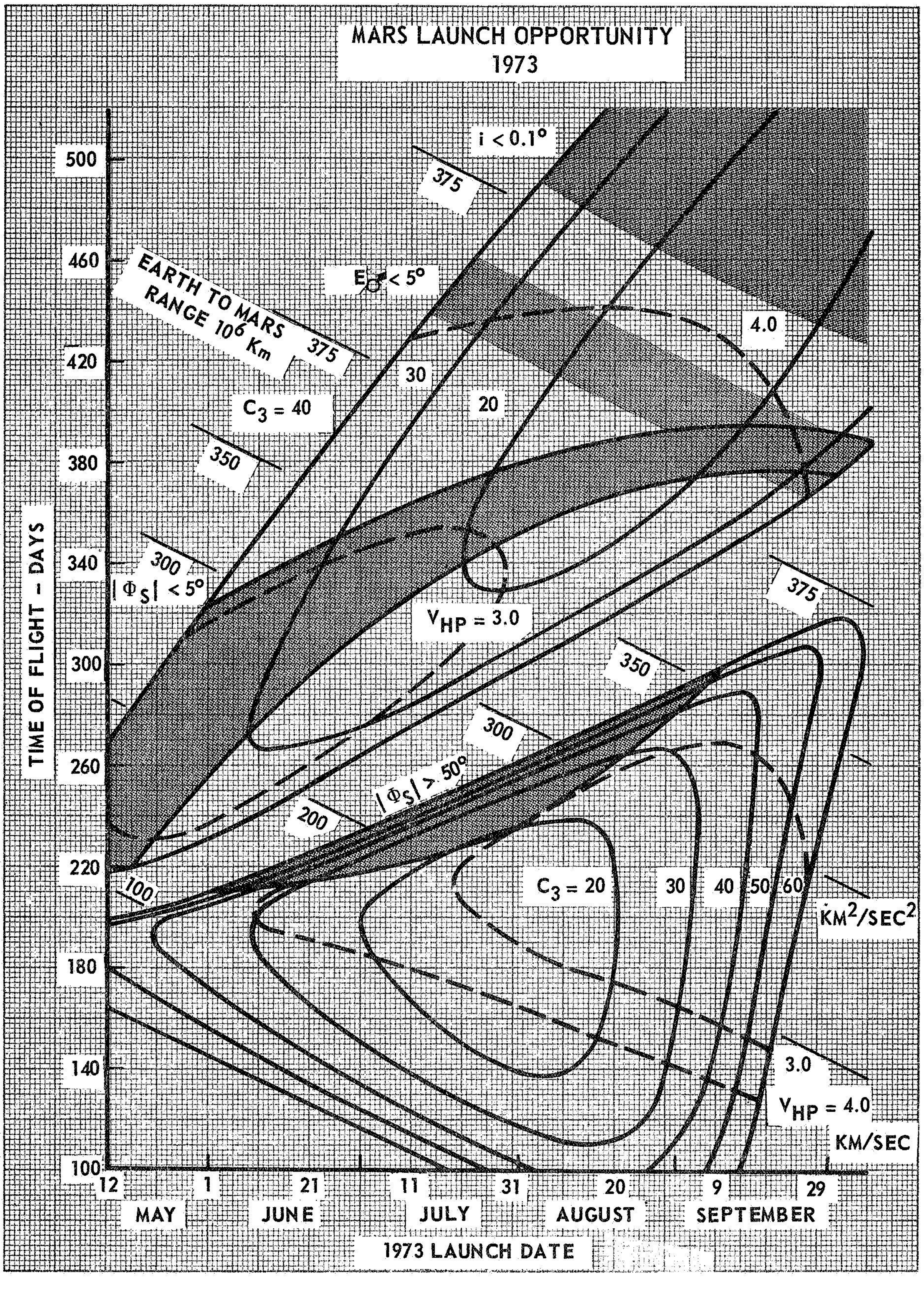


FIGURE 1.1-1



MARS LAUNCH OPPORTUNITY
1973
i < 0.1°
375
EARTH TO MARS
RANGE 10^6 Km
375
E♂ < 5°
4.0
30
20
C_3 = 40
350
300
$|\Phi_S| < 5°$
V_{HP} = 3.0
375
350
300
$|\Phi_S| > 50°$
200
100
C_3 = 20
30
40
50
60
KM^2/SEC^2
3.0
V_{HP} = 4.0
KM/SEC
TIME OF FLIGHT - DAYS
500
460
420
380
340
300
260
220
180
140
100
12
1
21
11
31
20
9
29
MAY
JUNE
JULY
AUGUST
SEPTEMBER
1973 LAUNCH DATE

FIGURE 1.1-2

TERMINAL DECELERATION SEQUENCE

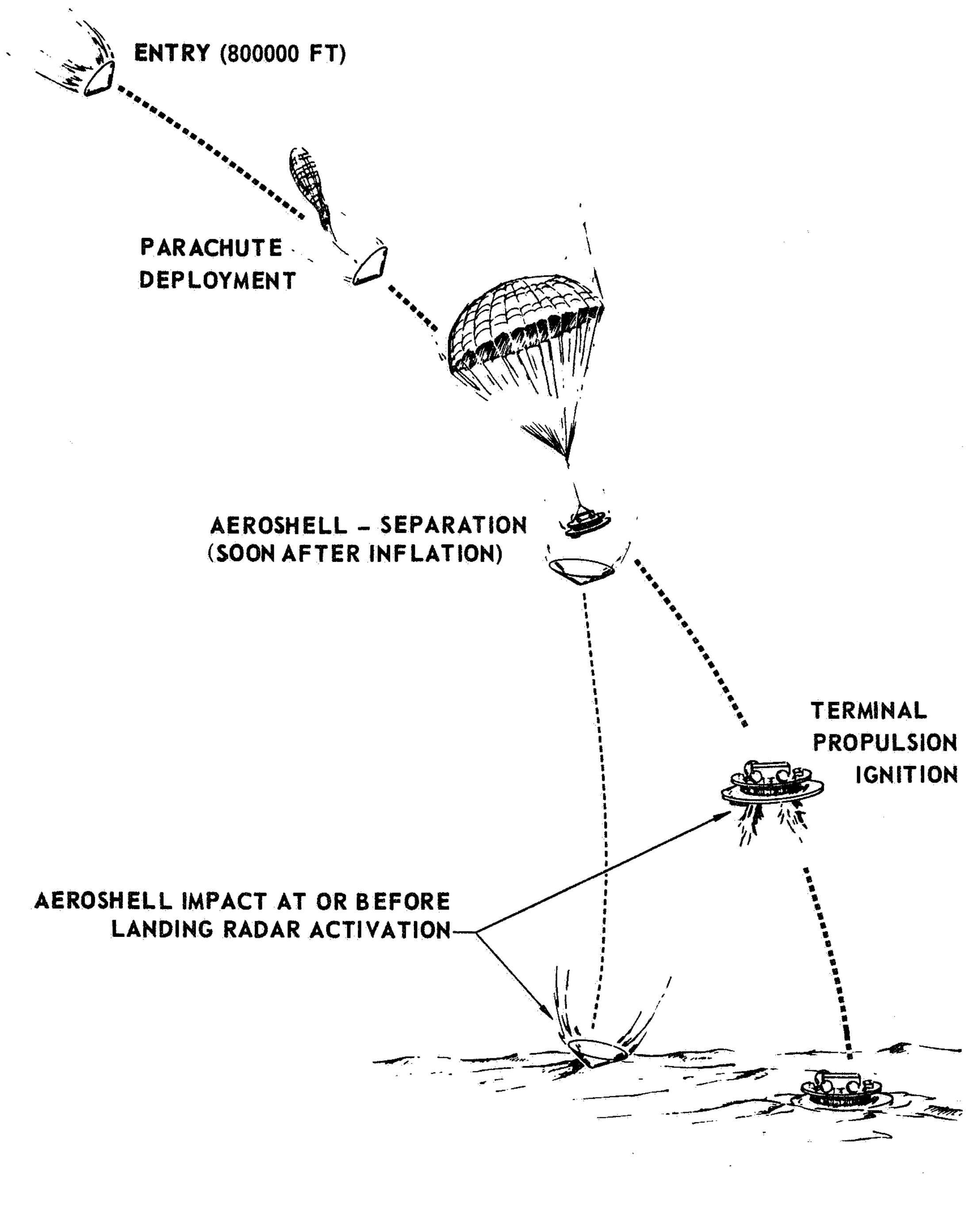


FIGURE 1.1–3

1.2 Capsule Design

The capsule consists of the surface payload and the systems which are necessary to deliver the payload to the Martian surface. The surface payload consists of the surface science complement, the communications equipment, power supply, thermal control, sequencer, and structure, wiring , and other equipment used to support the mission operation after landing on the Martian surface. The capsule terminology used in the report is presented in Figure 1.2-1.

1.2.1 DELIVERY SYSTEMS - The capsule characteristics are governed by several fundamental decisions. The entry body ballistic parameter (m/C_DA) must be small enough to provide sufficient deceleration to allow operation of auxiliary aerodynamic decelerators. Small m/C_DA's, however, provide longer atmospheric descent times, decreasing the time available for post-landed capsule-to-orbiter communications and increasing the total entry heating. Furthermore, small m/C_DA values are attained by using large aeroshell diameters, which may be incompatible with booster imposed limitations on the capsule dynamic envelope. The type of aerodynamic decelerator, if any, is related to the ballistic parameter; conservative decelerators demand lower ballistic parameters. The selection of monopropellant (simpler) or bipropellant (lighter) terminal propellant systems influences the size and cost of the capsule.

Ballistic parameters of about .25 slugs/sq ft are suitable for use with parachutes deployed at an altitude of 23 000 feet. This value can be increased by deploying the parachute at a lower altitude, but this requires larger parachutes to accomplish the desired deceleration and results in a small increase in capsule cost. The reduction in aeroshell diameter which would accompany this increase may warrant this additional cost. A reduction of parachute deployment altitude from 23 000 feet to 14 000 feet would permit an increase of m/C_DA to .35 slugs/sq ft and a 20% reduction in aeroshell diameter, as shown in Figure 1.2-2. However, additional parachute costs offset the effect of reduced aeroshell and canister structure costs.

Use of an aerodynamic inflatable decelerator system (AIDS) rather than a parachute would allow an increase in ballistic parameter to .40 slugs/sq ft

CAPSULE TERMINOLOGY

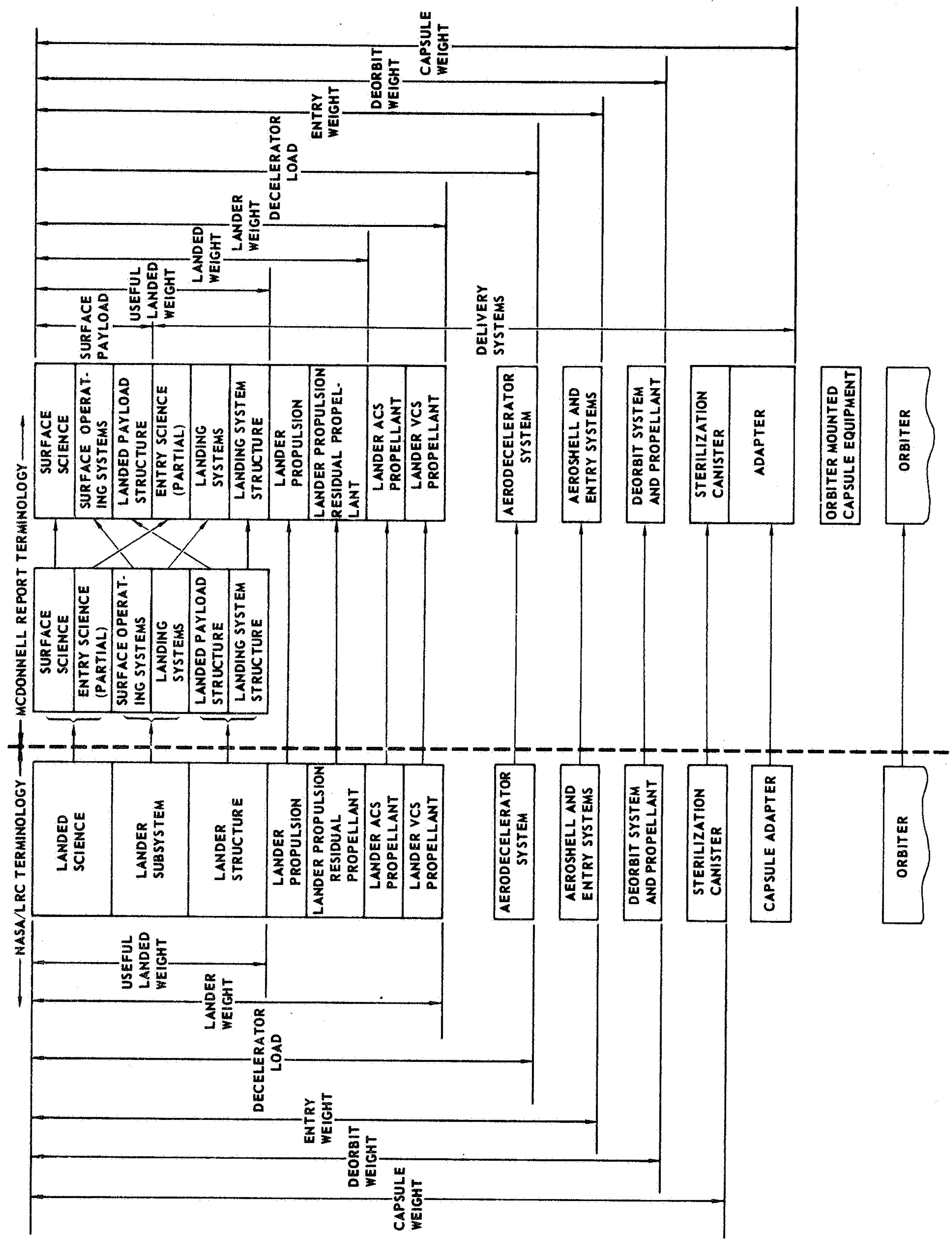


FIGURE 1.2-1

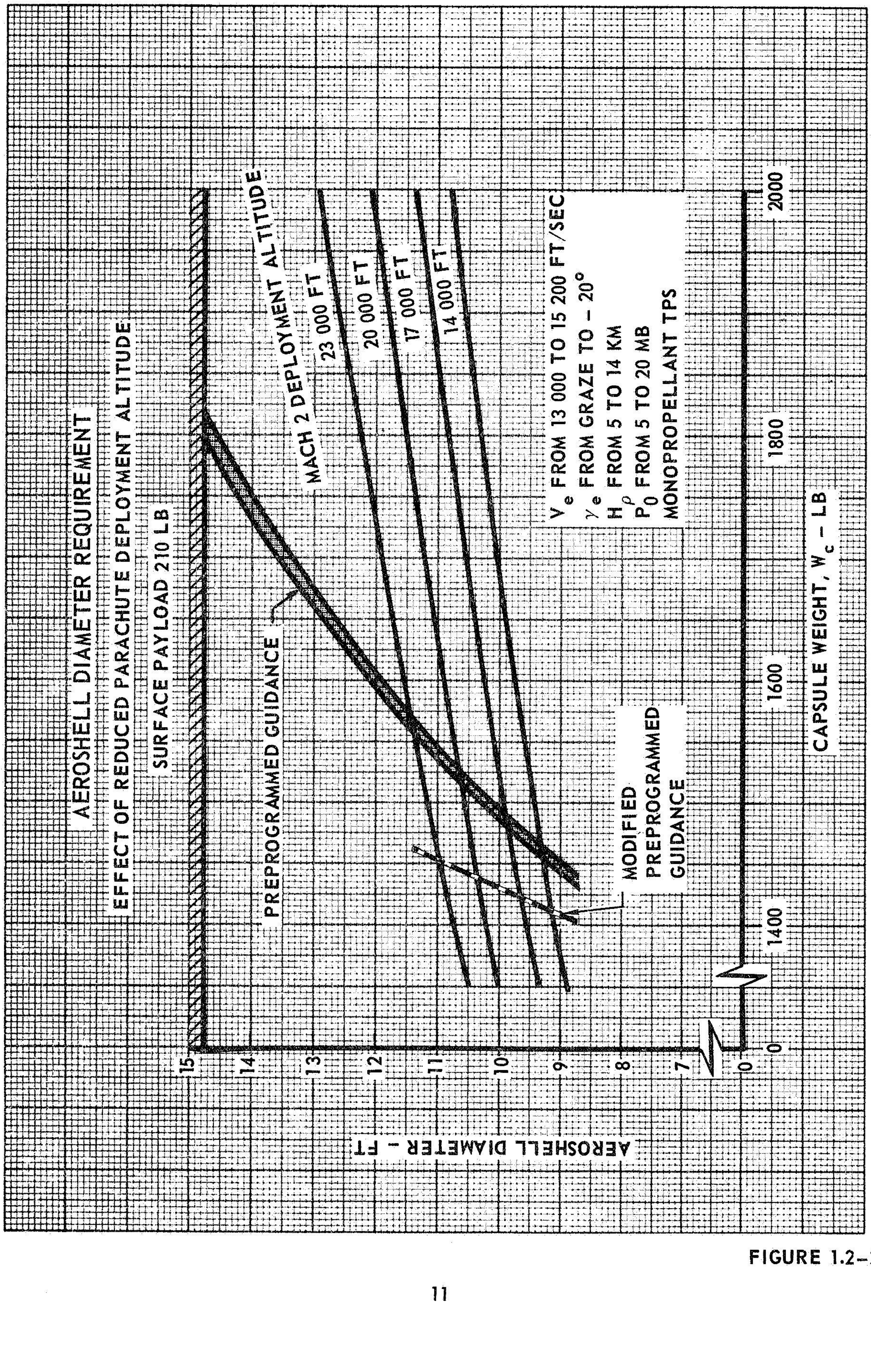

AEROSHELL DIAMETER REQUIREMENT
EFFECT OF REDUCED PARACHUTE DEPLOYMENT ALTITUDE
SURFACE PAYLOAD 210 LB
PREPROGRAMMED GUIDANCE
MODIFIED PREPROGRAMMED GUIDANCE
MACH 2 DEPLOYMENT ALTITUDE
23 000 FT
20 000 FT
17 000 FT
14 000 FT
V_e FROM 13 000 TO 15 200 FT/SEC
γ_e FROM GRAZE TO − 20°
H_ρ FROM 5 TO 14 KM
P_0 FROM 5 TO 20 MB
MONOPROPELLANT TPS
AEROSHELL DIAMETER - FT
15
14
13
12
11
10
9
8
7
0
CAPSULE WEIGHT, W_c − LB
0
1400
1600
1800
2000

FIGURE 1.2-2

and a reduction of aeroshell diameter from 11.4 feet to about 9 feet for a capsule with a 210 pound surface payload; see Figure 1.2-3. Although the AIDS would require more development and be more costly than a parachute for the same lander, the aeroshell and sterile canister size and cost would be significantly reduced.

Bipropellant terminal propulsion systems lead to lighter total capsules than monopropellant systems, but the cost of capsules using monopropellant terminal propulsion systems is less. For capsules weighing more than 3000 lb, the required development of a throttleable monopropellant engine with thrust of more than 1000 lb thrust presents sufficient development risk to offset this factor and supports selection of bipropellant systems.

1.2.2 SURFACE PAYLOAD - Within the surface payload, the paramount considerations are the science payload, data transmission mode, and surface mission duration. The choice of power system is closely related to the mission duration and to the choice of relay-via-orbiter (VHF) or direct-to-Earth (S-band) transmission mode.

Communications by orbiter relay permits use of a lighter capsule than does transmission direct to Earth, primarily because of increased power requirements for long range transmission. For high rate S-band systems a steerable antenna would be required. For the relay mode, the capsule weight and cost is relatively insensitive to data rate for rates below 10^7 bits per day. At higher rates, more complex data storage and transmission equipment is required and weight increases rapidly as data rate increases, as shown in Figure 1.2-4. The relay mode, however, requires that the orbiter be available for relay. For long duration missions, this imposes constraints on the orbiter mission and makes capsule performance dependent on orbiter reliability and orbit predictability. Both of these factors tend to counter the weight and cost advantages of the relay mode. A suitable compromise is to use the relay mode during the earlier days of the long duration mission when high quantities of data will be produced by the facsimile camera, and to transmit other data via a low rate S-band during the remainder of the mission. Long duration missions will probably require use of a command link to provide suitable flexibility on surface mission operations. The characteristics of the surface

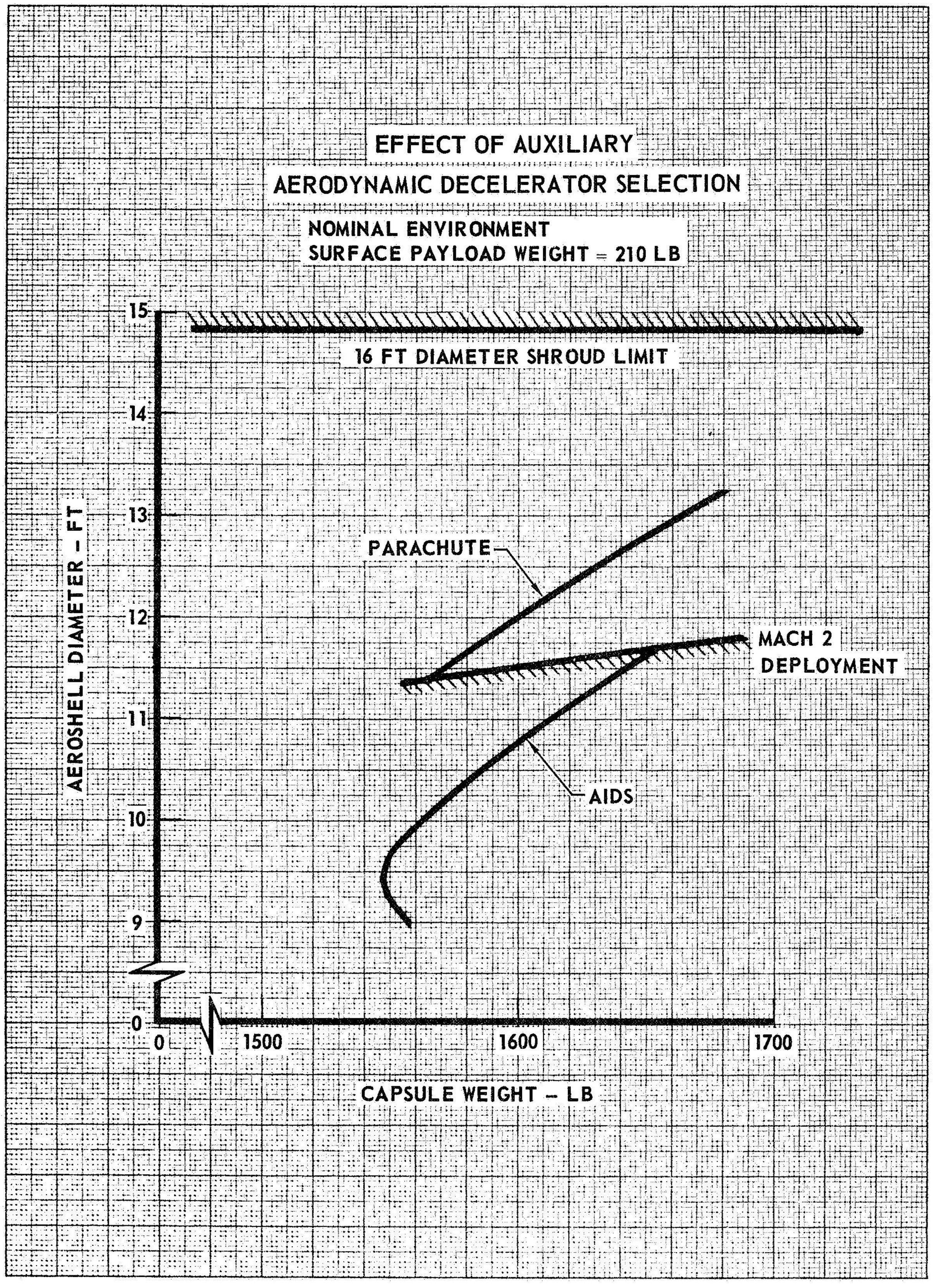

EFFECT OF AUXILIARY
AERODYNAMIC DECELERATOR SELECTION
NOMINAL ENVIRONMENT
SURFACE PAYLOAD WEIGHT = 210 LB
16 FT DIAMETER SHROUD LIMIT
PARACHUTE
MACH 2
DEPLOYMENT
AIDS
AEROSHELL DIAMETER – FT
15
14
13
12
11
10
9
0
0
1500
1600
1700
CAPSULE WEIGHT – LB

FIGURE 1.2–3

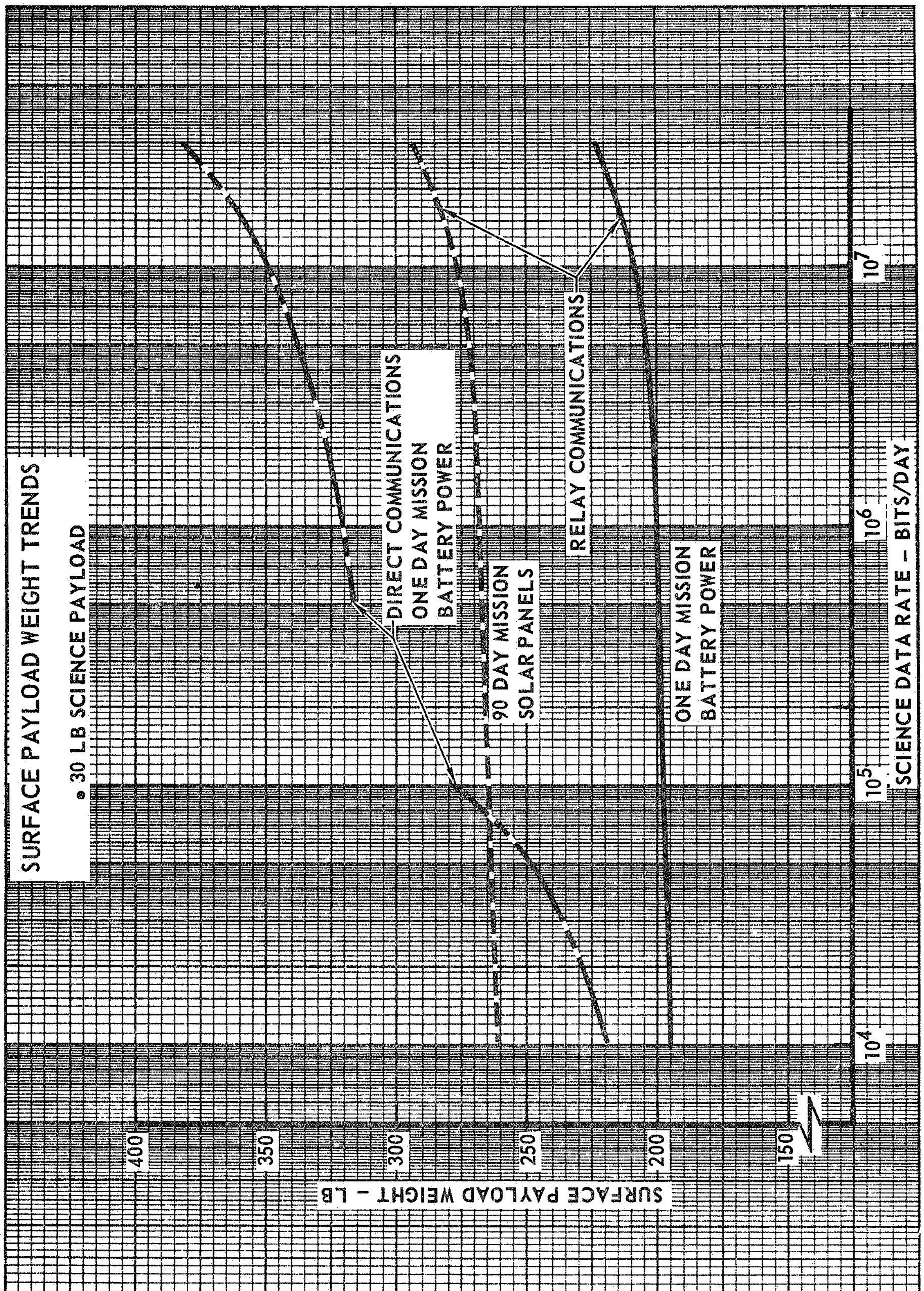

SURFACE PAYLOAD WEIGHT TRENDS
• 30 LB SCIENCE PAYLOAD
DIRECT COMMUNICATIONS
ONE DAY MISSION
BATTERY POWER
90 DAY MISSION
SOLAR PANELS
RELAY COMMUNICATIONS
ONE DAY MISSION
BATTERY POWER
SURFACE PAYLOAD WEIGHT – LB
150
200
250
300
350
400
10^4
10^5
10^6
10^7
SCIENCE DATA RATE – BITS/DAY

FIGURE 1.2-4

science complement and the associated data rates size the surface payload and therefore the total capsule.

Batteries are the most economical power supply for surface missions of a few days duration. For longer missions, a regenerative power supply must be provided. Both solar cells and radioisotope thermoelectric generators are suitable candidates. Because of the relative development status and lead time requirements, the solar cells have received emphasis in this study. Although they provide power for long periods, they are subject to degradation because of sand and dust deposits and cloud coverage, and their output is dependent on orientation and distance to the sun. As shown in Figure 1.2-5, approximately 20 square feet of solar panels are required for capsules with relay communications. This is small enough to allow fixed mounting on the surface of the lander.

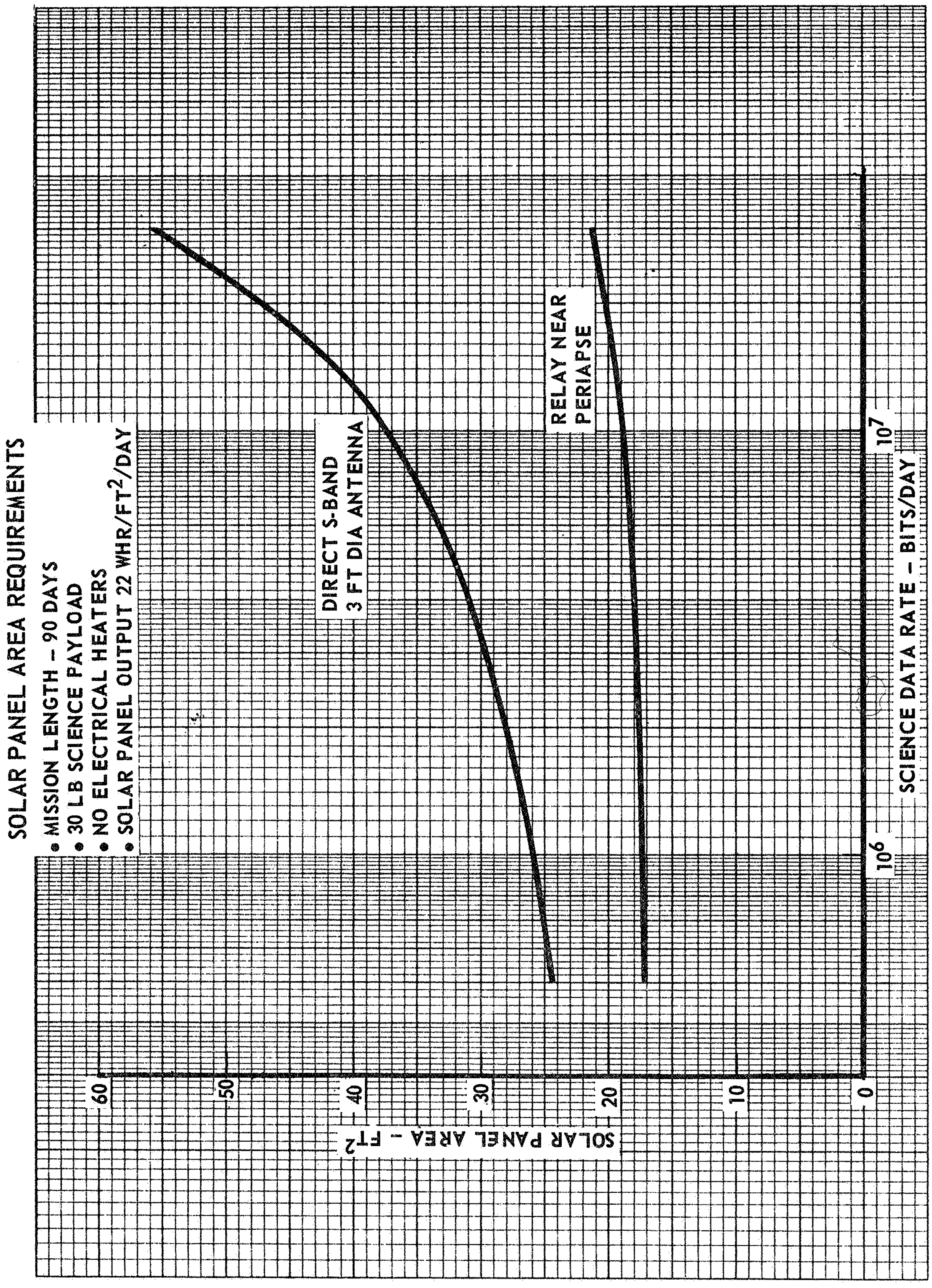
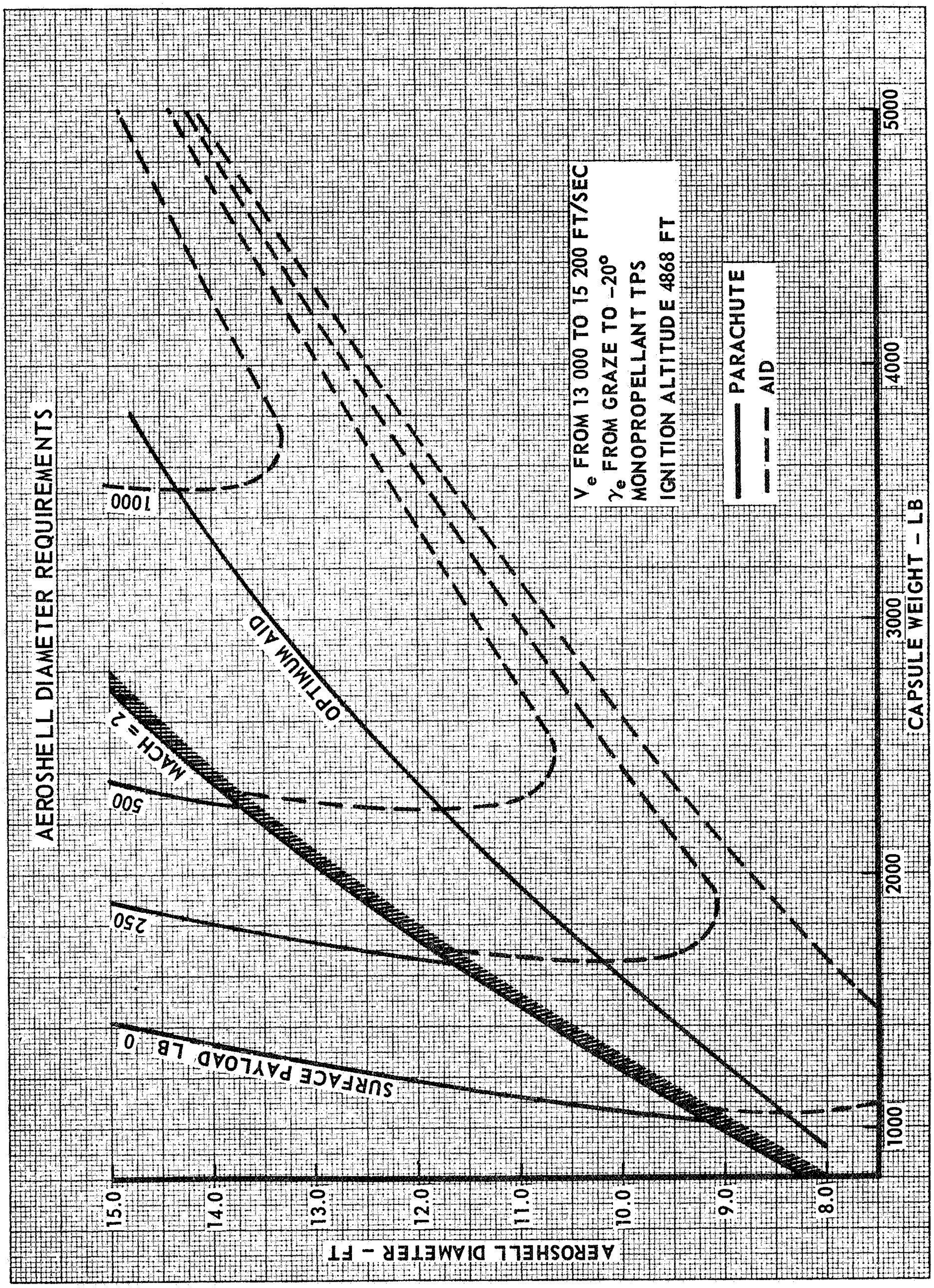

SOLAR PANEL AREA REQUIREMENTS
• MISSION LENGTH – 90 DAYS
• 30 LB SCIENCE PAYLOAD
• NO ELECTRICAL HEATERS
• SOLAR PANEL OUTPUT 22 WHR/FT^2/DAY
DIRECT S-BAND
3 FT DIA ANTENNA
RELAY NEAR
PERIAPSE
SOLAR PANEL AREA – FT^2
60
50
40
30
20
10
0
10^6
10^7
SCIENCE DATA RATE – BITS/DAY

FIGURE 1.2–5

1.3 Capsule Size and Weight

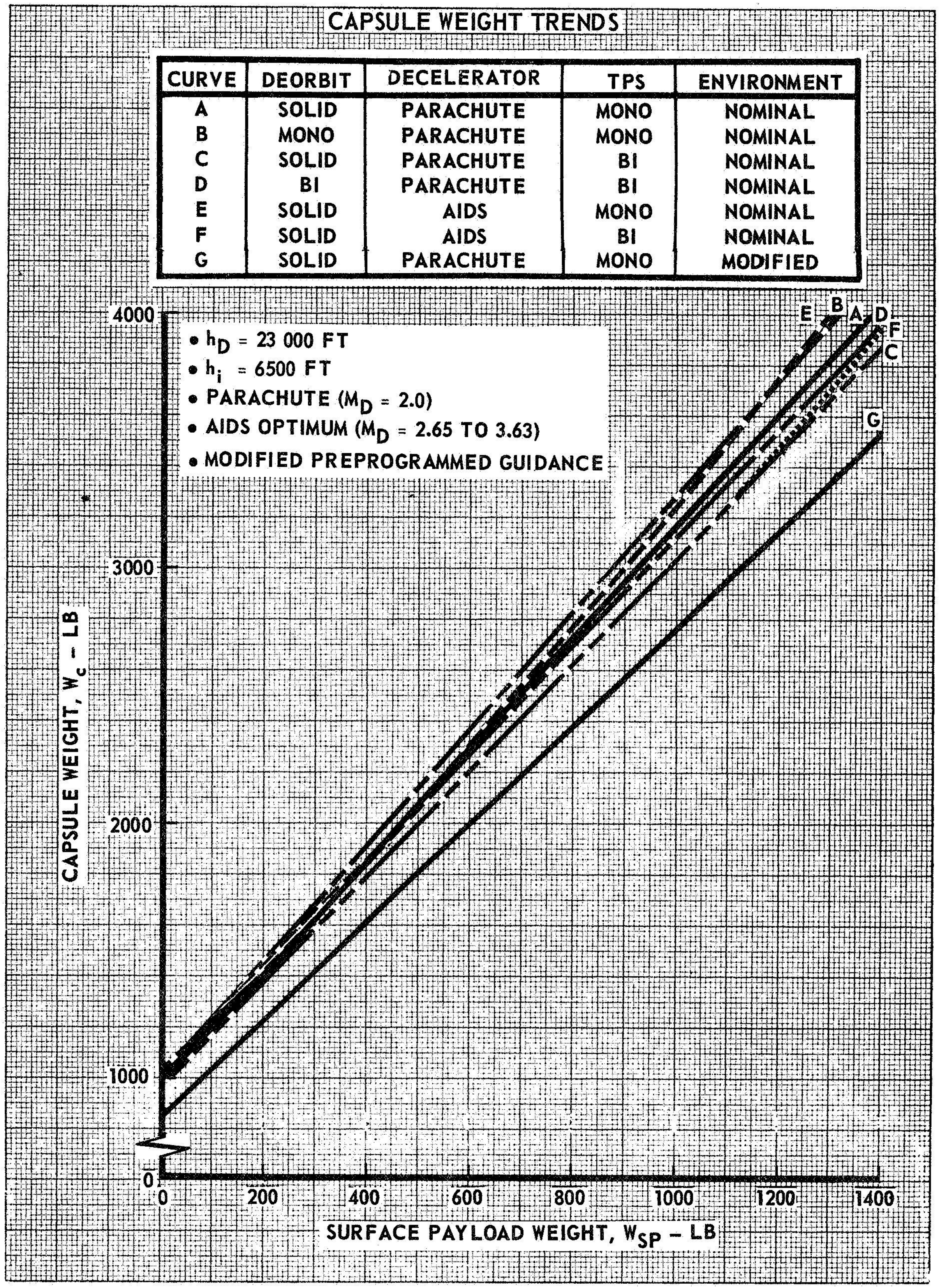
Capsule size and weight depend on the surface payload and the systems used for delivering the surface payload. Both size (aeroshell diameter is a convenient measure of capsule size) and weight increase with surface payload weight. However, there is some flexibility for selecting capsule weight and size for a given surface payload, as shown in Figure 1.3-1. The latitude allowed in making such a trade is limited by constraints such as the maximum aeroshell diameter and minimum $m/C_D A$, as shown in Figure 1.3-1. Differences in delivery systems selections vary the relationship of capsule and surface payload weight as shown in Figure 1.3-2 for a class of capsules with fixed ballistic parameters compatible with using parachutes. A rough approximation to the capsule weight can be obtained by noting that the capsule weights presented in Figure 1.3-2 are between 2.0 and 2.3 times the surface payload weight plus a weight of 1000 lb. More precise values for these and other weight growth trade factors are presented in Table 1.3-1 for systems that satisfy the study constraints. Aeroshell diameter trends are presented in Figure 1.3-3.

Surface payload weight for the minimum mission is between 200 and 325 lb depending on the type of support systems selected. For long duration missions using solar cells and relay communications, the surface payload weight is increased from slightly more than 200 lb to slightly more than 300 lb.



AEROSHELL DIAMETER REQUIREMENTS
AEROSHELL DIAMETER - FT
CAPSULE WEIGHT - LB
15.0
14.0
13.0
12.0
11.0
10.0
9.0
8.0
1000
2000
3000
4000
5000
SURFACE PAYLOAD LB 0
250
500
1000
MACH = 2
OPTIMUM AID
V_e FROM 13 000 TO 15 200 FT/SEC
γ_e FROM GRAZE TO -20°
MONOPROPELLANT TPS
IGNITION ALTITUDE 4868 FT
PARACHUTE
AID

FIGURE 1.3-1


CAPSULE WEIGHT TRENDS

CURVE	DEORBIT	DECELERATOR	TPS	ENVIRONMENT
A	SOLID	PARACHUTE	MONO	NOMINAL
B	MONO	PARACHUTE	MONO	NOMINAL
C	SOLID	PARACHUTE	BI	NOMINAL
D	BI	PARACHUTE	BI	NOMINAL
E	SOLID	AIDS	MONO	NOMINAL
F	SOLID	AIDS	BI	NOMINAL
G	SOLID	PARACHUTE	MONO	MODIFIED

• h_D = 23 000 FT
• h_i = 6500 FT
• PARACHUTE (M_D = 2.0)
• AIDS OPTIMUM (M_D = 2.65 TO 3.63)
• MODIFIED PREPROGRAMMED GUIDANCE
E
B
A
D
F
C
G
CAPSULE WEIGHT, W_c – LB
4000
3000
2000
1000
0
0
200
400
600
800
1000
1200
1400
SURFACE PAYLOAD WEIGHT, W_{SP} – LB

FIGURE 1.3–2

TABLE 1.3-1

CAPSULE WEIGHT RELATIONSHIPS

GROUP IDENTIFICATION	PROPULSION SYSTEMS – AERODECELERATOR SYSTEM COMBINATIONS	ENVIRONMENT	WEIGHT EXCHANGE EQUATIONS
A	SOLID PROPELLANT DEORBIT PARACHUTE MONOPROPELLANT TPS	NOMINAL	$W_c = 980 + 2.19\ W_{SP}$ $W_L = 528 + 1.40\ W_{SP}$ $W_{D.L.} = -5 + 1.05\ W_L$ $W_E = 70 + 1.22\ W_{D.L.}$ $W_C = 80 + 1.22\ W_E$ $D_{A/S} \approx 5.54 + .00467\ W_E$
B	MONOPROPELLANT DEORBIT PARACHUTE MONOPROPELLANT TPS	NOMINAL	$W_c = 1015 + 2.27\ W_{SP}$
C	SOLID PROPELLANT DEORBIT PARACHUTE BIPROPELLANT TPS	NOMINAL	$W_c = 973 + 2.07\ W_{SP}$
D	BIPROPELLANT DEORBIT PARACHUTE BIPROPELLANT TPS	NOMINAL	$W_c = 1015 + 2.11\ W_{SP}$
E	SOLID PROPELLANT DEORBIT AIDS MONOPROPELLANT TPS	NOMINAL	$W_c = 1000 + 2.16\ W_{SP}$ $W_L = 540 + 1.45\ W_{SP}$ $W_{DL} = -8 + 1.12\ W_L$ $W_E = 80 + 1.16\ W_{D.L.}$ $W_C = 110 + 1.17\ W_E$ $D_{A/S} \approx 5.85 + .00315\ W_E$
F	SOLID PROPELLANT DEORBIT AIDS BIPROPELLANT	NOMINAL	$W_c = 977 + 2.07\ W_{SP}$
G	SOLID PROPELLANT DEORBIT PARACHUTE MONOPROPELLANT TPS	MODIFIED	$W_c = 875 + 1.90\ W_{SP}$

NOTES: SP – SURFACE PAYLOAD
L – LANDER
DL – DECELERATOR LOAD
E – ENTRY WEIGHT
C – CAPSULE
A/S – AEROSHELL
W – WEIGHT–LB
D – DIAMETER–FT

AEROSHELL DIAMETER TRENDS

CURVE	DEORBIT	DECELERATOR	TPS	ENVIRONMENT
A	SOLID	PARACHUTE	MONO	NOMINAL
B	MONO	PARACHUTE	MONO	NOMINAL
C	SOLID	PARACHUTE	BI	NOMINAL
D	BI	PARACHUTE	BI	NOMINAL
E	SOLID	AIDS	MONO	NOMINAL
F	SOLID	AIDS	BI	NOMINAL
G	SOLID	PARACHUTE	MONO	MODIFIED

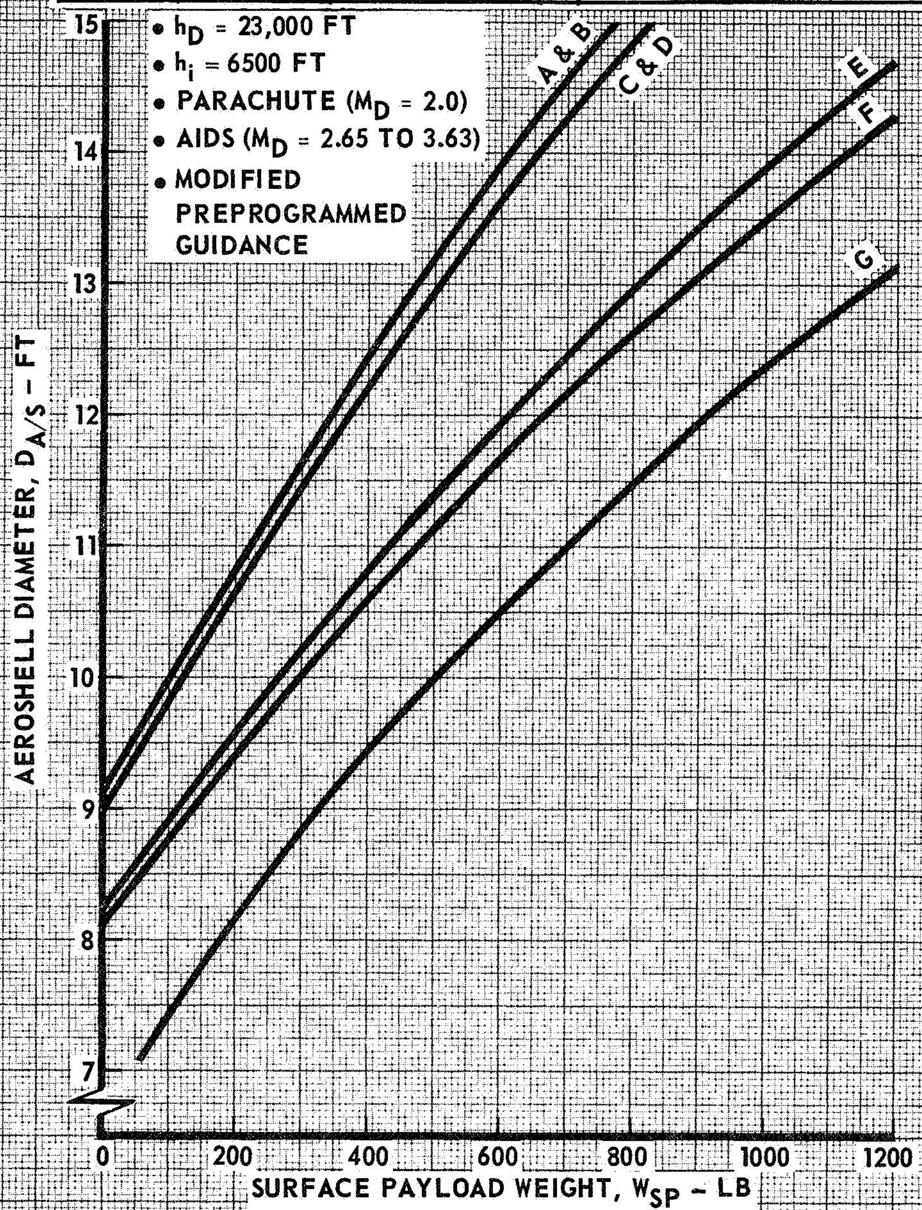


FIGURE 1.3–3

1.4 Capsule Systems

Parametric studies of the design and performance of the capsule systems were performed as the basis for study of the soft lander capsule. Systems which permit development for the 1973 launch opportunity were considered. The development status of major components is indicated in Table 1.4-1. In order to meet the 1973 launch schedule, early attention should be directed to the development of technology in the areas of sterilizable batteries and sterizable solid propellant rockets. The throttleable liquid propulsion system requires the longest lead time of the capsule systems.

1.4.1 SCIENCE - The characteristics of the science payload which will measure entry and surface atmospheric parameters, soil composition, and provide images of the surface after landing are presented in Table 1.4-2. Focus control will be required for the facsimile camera because the low light levels available on the Martian surface will require large lens diameters. The use of a counting system with the alpha spectrometer will minimize the quantity of data to be transmitted; the development of a new radioactive source technique will be required for the one day missions to keep the data acquisition time sufficiently short. No serious difficulties are anticipated in developing the instruments for the 1973 mission.

1.4.2 COMMUNICATIONS - Science and engineering data will be transmitted by a UHF system to the orbiter for relay to Earth during the descent and after landing. Antenna patterns, capsule-orbiter geometry, and reflected-signal interference are primary inputs to analysis of the descent communications requirements. Landed relay communication is influenced by orbiter characteristics and the data rate requirements. Transmitter energy required for alternative post-landing communication modes is compared in Figure 1.4-1. For extended missions the relay will be supplemented by two-way, S-band, direct-to-Earth communications.

1.4.3 POWER - Primary silver-zinc batteries will supply the electrical energy requirement for surface missions of a few days duration. Secondary silver-zinc batteries will be required in conjunction with a regenerative power source, such as a solar cell array or a radioisotope thermoelectric generator (RTG), for longer missions. Sterilization of these batteries is a

TABLE 1.4–1
SOFT LANDER SYSTEMS – DEVELOPMENT STATUS

SYSTEM • MAJOR ASSEMBLY MINOR ASSEMBLY	DEVELOPMENT STATUS EQUIVALENT HARDWARE	DEVELOPMENT STATUS PROPOSED SOFT LANDER HARDWARE	REMARKS
SCIENCE			
• FACSIMILIE CAMERA (LOW RESOLUTION)	SIMILAR HDW QUALIFIED	PROTOTYPE AVAILABLE	
• FACSIMILIE CAMERA (HIGH RESOLUTION)	NONE DEVELOPED	NO WORK TO BEGIN BEFORE PHASE C RFP	SOME ANALYSIS BEING DONE AT PHILCO
• PRESSURE TRANSDUCER (LANDED)	SIMILAR HDW FLOWN	STERILIZABLE HDW AVAILABLE	
• TEMPERATURE TRANSDUCER (LANDED)			
• WIND SENSOR (LANDED)	SIMILAR HDW IN DEVELOPMENT		
• HUMIDITY SENSOR (LANDED)			
• ALPHA SPECTROMETER		SEVERAL BREADBOARD MODELS IN WORK	DATA CONVERTER MUST BE DEVELOPED
• MASS SPECTROMETER (ENTRY)	SIMILAR HDW FLOWN		
• MOISTURE SENSOR (ENTRY)	IN DEVELOPMENT		
• TEMPERATURE SENSOR (ENTRY)		STERILIZABLE HDW AVAILABLE	
• ATMOSPHERE PRESSURE (ENTRY)	SIMILAR HDW FLOWN		
• PRESSURE SENSOR (AEROSHELL NOSE)		NO WORK WILL BEGIN PRIOR TO PHASE C RFP	SENSING ELEMENTS ARE MOST CRITICAL
• TRI-AXIS ACCELEROMETER			SERVO LOOP IS PROBABLY CRITICAL
COMMUNICATIONS			
• ANTENNA SYSTEM			
ORBITER UHF ANTENNA			REQUIREMENTS FOR LOW LOSS & LOW WEIGHT MAKE DIPLEXERS/CIRCULATORS MOST CRITICAL ANTENNA DEVELOPMENT ITEMS
CAPSULE UHF ANTENNA	SIMILAR ANTENNAS FLOWN ON SEVERAL PROGRAMS		
CAPSULE S-BAND ANTENNA			
• RADIO SYSTEM			
ORBITER UHF RECEIVER	PARTIALLY DEVELOPED BY PHILCO FOR VOYAGER		RECEIVER NOISE FIGURE IS CRITICAL. BIT SYNCHRONIZERS ARE NEW DEVELOPMENT.
CAPSULE ENTRY TRANSMITTER	SEVERAL SIMILAR TRANSMITTERS FLOWN		UHF TRANSMITTER FINAL STAGE POWER ARE CRITICAL BECAUSE OF LOW WEIGHT REQUIRED
CAPSULE RELAY TRANSMITTER			MFSK GENERATOR IS NEW DEVELOPMENT ITEM
CAPSULE S-BAND TRANSMITTER	SOME WORK BY JPL & PHILCO FOR VOYAGER		
• TELEMETRY SYSTEM			
ORBITER COMMUTATOR			
CAPSULE COMMUTATOR	SEVERAL FLOWN		PROGRAMMABLE MEMORY IS A PROVEN CONCEPT ONLY
CAPSULE SIGNAL PROCESSOR			
• COMMAND SYSTEM			
ORBITER DECODER	SEVERAL FLOWN		
CAPSULE RECEIVER DECODER			RECEIVER NOISE FIGURE WILL BE CRITICAL
• STORAGE SYSTEM			
ORBITER TAPE RECORDER	SEVERAL FLOWN		
CAPSULE CORE STACK			MEMORY STACK INTERCONNECTIONS ARE CRITICAL
POWER			
• SILVER-ZINC BATTERY	IN DEVELOPMENT AT DOUGLAS ASTROPOWER FOR NASA–LEWIS	PROTOTYPE AVAILABLE BY LATE 1969	CHEMICAL SYSTEM SENSITIVE TO STERILIZATION
• BATTERY CHARGER			
• CONVERT – REGULATOR	SEVERAL SIMILAR FLOWN	NO WORK TO BEGIN PRIOR TO PHASE C RFP	NO CRITICAL DEVELOPMENT IS FORECAST
• POWER SWITCHING/LOGIC UNIT			
• SOLAR CELL ARRAY			
GUIDANCE & CONTROL			
• INERTIAL MEASUREMENT UNIT			
GYROS	DEVELOPED BY HONEYWELL UNDER JPL CONTRACT	HONEYWELL GYROS APPLICABLE	
ACCELEROMETER	CHOSEN ACCELEROMETER	HAS ALREADY FLOWN	
• COMPUTER	SIMILAR MAGNETIC CORES FLOWN	NO WORK BEFORE PHASE C RFP	EXISTING HARDWARE MAY BE MODIFIED
• LANDING RADAR	LM LANDING RADAR	MINIMUM MODICATIONS TO LM LANDING RADAR	STERILIZATION, PLUME DAMAGE AND ALTERNATE ANTENNA CONFIGURATION ARE PROBLEMS.
• RADAR ALTIMETER	SIMILAR HDW FLOWN	USES EXISTING PULSE ALTIMETRY TECHNIQUES	NORMAL DEVELOPMENT AND TESTS REQUIRED
SEQUENCER			
• CAPSULE SEQUENCER/TEST PROGRAMMER	SIMILAR HDW FLOWN	NO WORK TO BEGIN PRIOR TO PHASE C RFP	SIMILAR UNITS HAVE BEEN STERILIZED
• SURFACE SEQUENCER & TIMER	SIMILAR HDW QUALIFIED		
THERMAL CONTROL			
• CAPSULE THERMAL CONTROL			
MULTILAYER INSULATION		STERILIZABLE INSULATION IS NOW AVAILABLE	
ELECTRIC HEATERS	SIMILAR EQUIPMENT FLOWN ON SEVERAL PROGRAMS	NO WORK WILL BEGIN PRIOR TO PHASE C RFP	LONG TIME IS REQUIRED FOR DEVELOPMENT TESTS
THERMOSTATS			
EQUIPMENT INSULATION			
SPACE STABLE COATINGS		COATINGS ARE AVAILABLE	
• LANDER THERMAL CONTROL			
INSULATION SYSTEM	SIMILAR HDW FLOWN		
ISOTOPE HEATERS	SIMILAR HDW QUALIFIED	PRODUCTION HEATER AVAILABLE	
HEATER DEPLOYMENT ASSEMBLIES	SIMILAR HDW QUALIFIED		
ELECTRIC HEATERS			LONG TIME IS REQUIRED FOR DEVELOPMENT TESTS
THERMOSTATS			
HEAT SINK MATERIAL	SIMILAR HDW FLOWN	STERILIZABLE PRODUCTION MATERIAL AVAILABLE	
PACKAGED PHASE CHANGE MATERIAL			
SPACE/MARS STABLE COATINGS	SIMILAR HDW QUALIFIED	COATINGS ARE AVAILABLE	

TABLE 1.4-1
SOFT LANDER SYSTEMS – DEVELOPMENT STATUS (Continued)

SYSTEM • MAJOR ASSEMBLY MINOR ASSEMBLY	DEVELOPMENT STATUS EQUIVALENT HARDWARE	 PROPOSED SOFT LANDER HARDWARE	REMARKS
PROPULSION			
• DEORBIT ROCKET			
PROPELLANT GRAIN	SIMILAR HDW IN DEVELOPMENT	PROTOTYPE BEING DEVELOPED	
CASE	SIMILAR HDW FLOWN	STERILIZABLE HDW AVAILABLE	
INSULATION (INTERNAL)	NO SIMILAR HDW	NO WORK BEFORE PHASE C	CURRENT INSULATION SYSTEMS ARE INADEQUATE FOR STERILIZABLE SYSTEMS
NOZZLE ASSEMBLY	SIMILAR HDW FLOWN	STERILIZABLE HDW AVAILABLE	
IGNITER	SIMILAR HDW IN DEVELOPMENT	PROTOTYPE BEING DEVELOPED	
• ATTITUDE CONTROL (COLD GAS)			
PRESSURANT TANK		STERILIZABLE HDW AVAILABLE	
REGULATOR	SIMILAR HDW FLOWN	PRODUCTION HDW AVAILABLE	
VALVES		PROTOTYPE IN DEVELOPMENT	
THRUSTERS		PRODUCTION HDW AVAILABLE	
• TERMINAL PROPULSION (MONO)			
PRESSURANT TANK	SIMILAR HDW FLOWN	STERILIZABLE HDW AVAILABLE	MORE WORK IS REQUIRED ON PROPELLANT MATERIAL COMPATIBILITY. 300 LB FIXED THRUST ENGINE IS IN DEVELOPMENT AND 10:1 THROTTLE-ABLE, 1000 LB PROTOTYPE ENGINE IS TO BE FUNDED. POTENTIAL PROBLEMS INCLUDE HEAT SOAK-BACK AND CHUGGING INSTABILITY
PROPELLANT TANK		NO WORK BEFORE PHASE C	
VALVES		PROTOTYPE IN WORK	
ENGINE ASSEMBLY	SIMILAR HDW IN DEVELOPMENT	PRODUCTION HDW AVAILABLE BY PHASE C	
AUXILLIARY AERO-DECELERATOR			
• CATAPULT ASSEMBLY			
CATAPULT	SIMILAR HDW FLOWN		
CANISTER			
DEPLOYMENT BAG	NO SIMILAR HDW DEVELOPED	NO WORK TO BEGIN PRIOR TO PHASE C RFP	INTERNAL COMPARTMENTATION LACING REQUIRES DEVELOPMENT
• REEFING LINE CUTTER	SIMILAR HDW FLOWN		
• PARACHUTE	SIMILAR HDW IN DEVELOPMENT		ADDITIONAL TESTING BY NASA UNDERWAY
AEROSHELL			
• STRUCTURE			
NOSE SECTION	SIMILAR HDW FLOWN		
CONICAL SECTION	SIMILAR HDW FLOWN		
• HEAT SHIELD			HFC DEVELOPMENT TESTS AND NON-DESCTRUCTIVE BOND TESTING ARE REQUIRED
NOSE SECTION	SIMILAR HDW QUALIFIED	NO WORK TO BEGIN PRIOR TO PHASE C RFP	MDC S–20T WITH HONEYCOMB REQUIRES DEVELOPMENT TESTING WITH SELECTED STRUCTURE
CONICAL SECTION	SIMILAR HDW FLOWN		COATED SILICA CLOTH MUST BE DEVELOPED FOR SPECIFIC ENTRY ENVIRONMENT
AFT THERMAL CURTAIN	SIMILAR HDW FLOWN		
CUTOUT ASSEMBLIES (WINDOWS)	SIMILAR HDW FLOWN		
• SEPARATION SYSTEM			
LANDING SYSTEM			
• DEORBIT MOTOR PARACHUTE SUPPORT ASSEMBLY			SEPARATION ASSEMBLY – EXPLOSIVE BOLTS TESTING REQUIRED
• SOLAR PANEL SUPPORT ASSEMBLY	SIMILAR HDW FLOWN		
• SURFACE PAYLOAD STRUCTURE ASSEMBLY			EQUIPMENT COLD PLATE WITH PHASE CHANGE MATERIAL IS MOST CRITICAL DEVELOPMENT ITEM. HEAT SHORTS THRU INSULATION SUPPORT STRUCTURE MUST BE MINIMIZED.
• EQUIPMENT MODULE ASSEMBLY			INSULATION SUPPORT STRUCTURE ASSEMBLY HEAT SHORTS MUST BE MINIMIZED.
• DEPLOYMENT MECHANISMS			UNLOCKING MECHANISMS REQUIRE POST-STERILIZATION TESTING.
ATMOSPHERIC SENSORS		NO WORK TO BEGIN PRIOR TO PHASE C RFP	
FACSIMILE CAMERA	SIMILAR EQUIPMENT FLOWN ON SEVERAL PROGRAMS		
UHF ANTENNA			
ALPHA-SPECTROMETER			
• BASE PLATFORM			
• LANDING FOOTPAD			
• PYROTECHNIC BOLT CUTTER ON STABILIZING LESS			
• ATTENUATOR RING			DEVELOPMENT OF FABRICATION PROCEDURE IS REQUIRED
• TENSION CABLE ASSEMBLY	SIMILAR HDW FLOWN		
CANISTER & ADAPTER			
• STRUCTURE			
FWD & AFT CANISTER ASSEMBLY	SIMILAR HDW FLOWN	NO WORK TO BEGIN PRIOR TO PHASE C RFP	
TRUSS ASSEMBLY			
• SEPARATION SYSTEM CANISTER	SIMILAR HDW (SKIN CUTTING) IN QUAL TESTS ON MOL		CONFINED EXPLOSIVE SEPARATION DEVICE REQUIRES MINOR DEVELOPMENT
ADAPTER/LANDER	SIMILAR HDW FLOWN	POSSIBLE USE OF OFF SHELF BOLTS	MINIMUM DEVELOPMENT IS REQUIRED
• PRESSURIZATION & VENTING SYSTEM			
VENT VALVES			
RELIEF VALVES	SIMILAR HDW FLOWN	NO WORK TO BEGIN PRIOR TO PHASE C RFP	MINIMUM DEVELOPMENT IS REQUIRED
PURGE & EVACUATION VALVES			
FILTERS	NO SIMILAR HDW DEVELOPED		
DIFFERENTIAL PRESSURE SWITCH	SIMILAR HDW FLOWN		

TABLE 1.4–2

SCIENCE PAYLOADS

	WEIGHT LB.	POWER WATTS	DIRECT DATA BITS/DAY	RELAY DATA BITS/DAY
LANDED SCIENCE				
CAMERAS	10	15	–	10^7
ATMOSPHERIC PACKAGE	10	7	10^3	9.4×10^3
ALPHA SPECTROMETER	10	2	–	8.4×10^3
MASS SPECTROMETER	(9)	(8)	–	(1.6×10^3)
TOTAL	30	24	10^3	17.8×10^3 BITS/DAY PLUS IMAGES

	WEIGHT LB	POWER WATTS	AVERAGE DATA RATE BITS/SEC
ENTRY SCIENCE			
MASS SPECTROMETER	9	8	80
ACCELEROMETER	2	4	150
PRESSURE SENSORS	5	3	48
TEMPERATURE SENSORS	1	0.02	16
HUMIDITY	1	0.02	8
MARGIN	5		
TOTAL	23	15	302

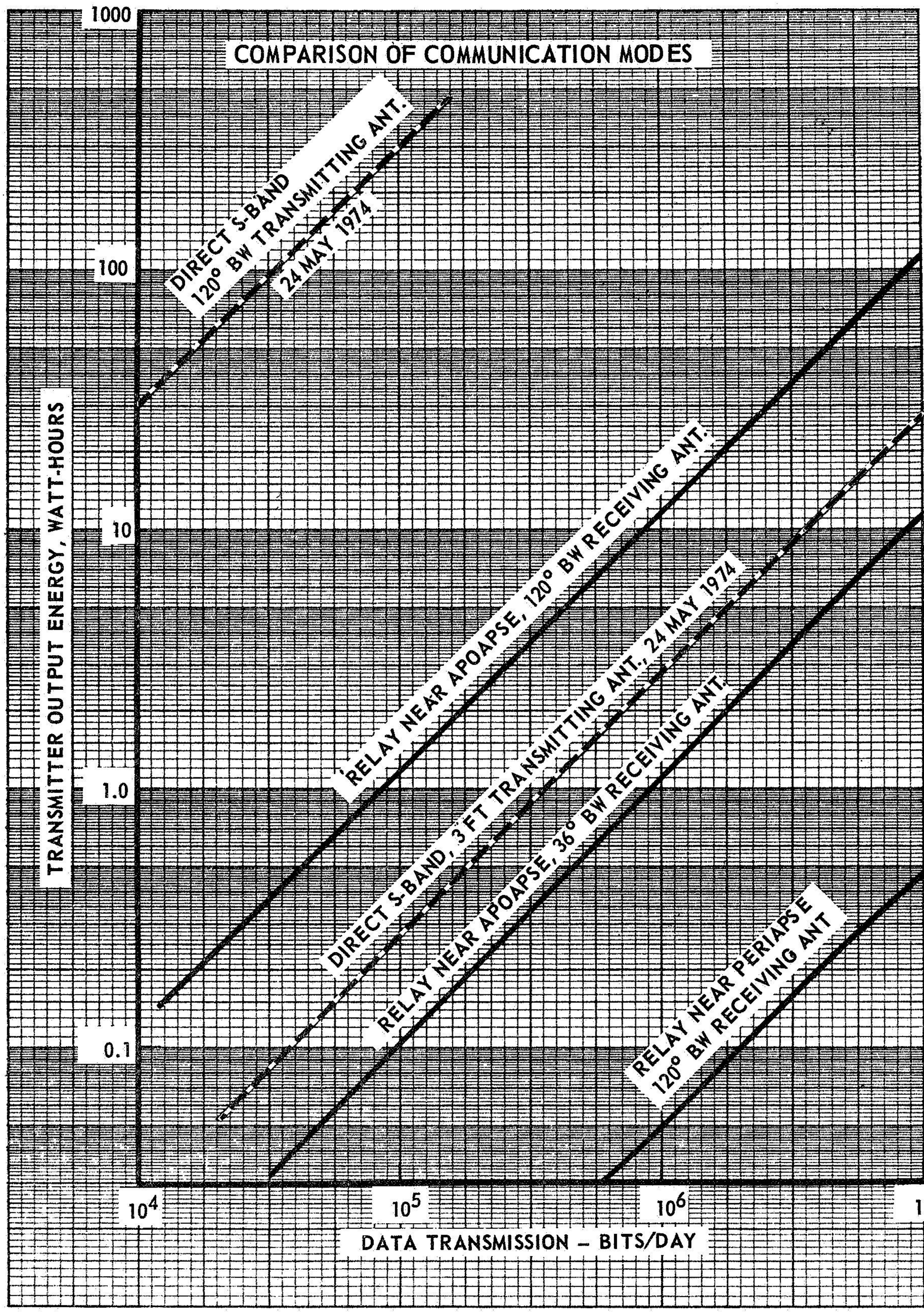

COMPARISON OF COMMUNICATION MODES
1000
100
10
1.0
0.1
DIRECT S-BAND 120° BW TRANSMITTING ANT. 24 MAY 1974
RELAY NEAR APOAPSE, 120° BW RECEIVING ANT.
DIRECT S-BAND, 3 FT TRANSMITTING ANT., 24 MAY 1974
RELAY NEAR APOAPSE, 36° BW RECEIVING ANT.
RELAY NEAR PERIAPSE 120° BW RECEIVING ANT
TRANSMITTER OUTPUT ENERGY, WATT-HOURS
10^4
10^5
10^6
10^7
DATA TRANSMISSION – BITS/DAY

FIGURE 1.4–1

critical problem and continued, concentrated investigation and development is required to provide qualified hardware for a 1973 launch. Figure 1.4-2 presents a projection of energy density for wet, heat-sterilizable, silver-zinc batteries based on continuing progress.

For extended operation of a Martian lander, a solar cell array offers an attractive power source that is not weight-sensitive to total energy output. However, solar cell output is affected by array shape, landing site location, landed vehicle attitude, and Martian environment. Figure 1.4-3 shows the effect of landing date and landing site latitude on solar panel performance. The uncertainties in environmental conditions, particularly cloud cover and wind blown dust, require that conservative estimates be used in specifying solar array size requirements.

An RTG is a suitable power source for extended missions on the Martian surface because of its inherent long life, reliability, and relative insensitivity to the local environment. Very long lead time is required to design, develop, produce, and qualify an RTG. Even with an early start on needed modifications, only the SNAP-19 and SNAP-27 programs have progressed sufficiently to meet a 1973 launch date.

1.4.4 GUIDANCE AND CONTROL - The guidance and control system consists of the attitude, deorbit, and velocity control electronics; landing radar; and radar altimeter. This equipment is used to sense and control vehicle attitude, attitude rate, acceleration, range, and velocity from capsule separation to touchdown. Primary attitude, deorbit, and velocity control design considerations include the deorbit thrust direction pointing, deorbit velocity increment, roll reference, and velocity direction calculation accuracies. Primary contributors to the landing radar and radar altimeter requirements analysis include near-zero doppler performance, anticipated attitude and attitude rate extremes, potential post-separation aeroshell interference, antenna patterns, surface topography, and backscatter characteristics. The guidance and control system is based on existing equipments and techniques. However, the necessary modifications to existing equipment, required testing, and complexity of this system combine to make it a pacing item with regard to total capsule development time.

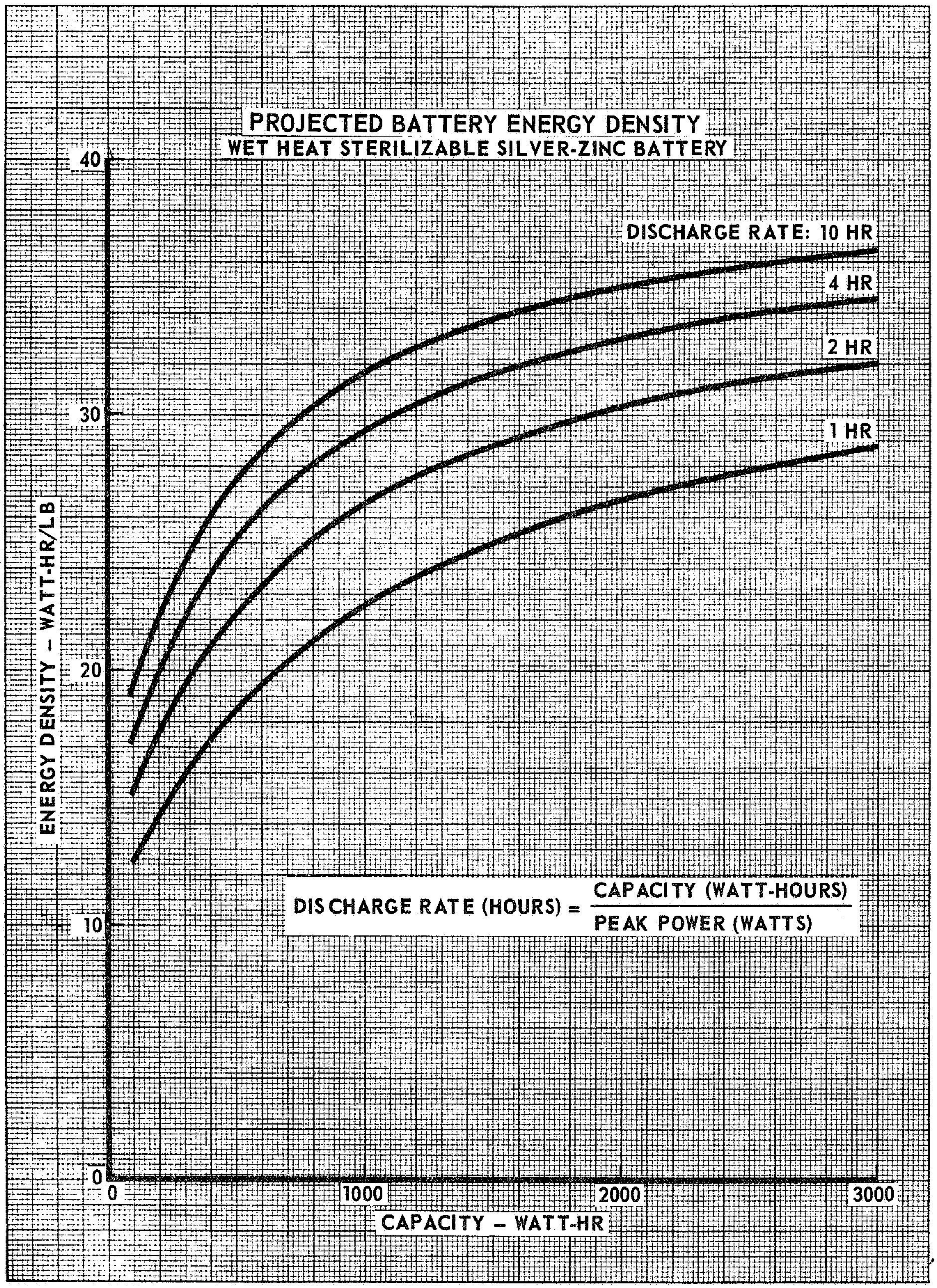

PROJECTED BATTERY ENERGY DENSITY
WET HEAT STERILIZABLE SILVER-ZINC BATTERY
DISCHARGE RATE: 10 HR
4 HR
2 HR
1 HR
DISCHARGE RATE (HOURS) = CAPACITY (WATT-HOURS) / PEAK POWER (WATTS)
ENERGY DENSITY – WATT-HR/LB
40
30
20
10
0
0
1000
2000
3000
CAPACITY – WATT-HR

FIGURE 1.4–2

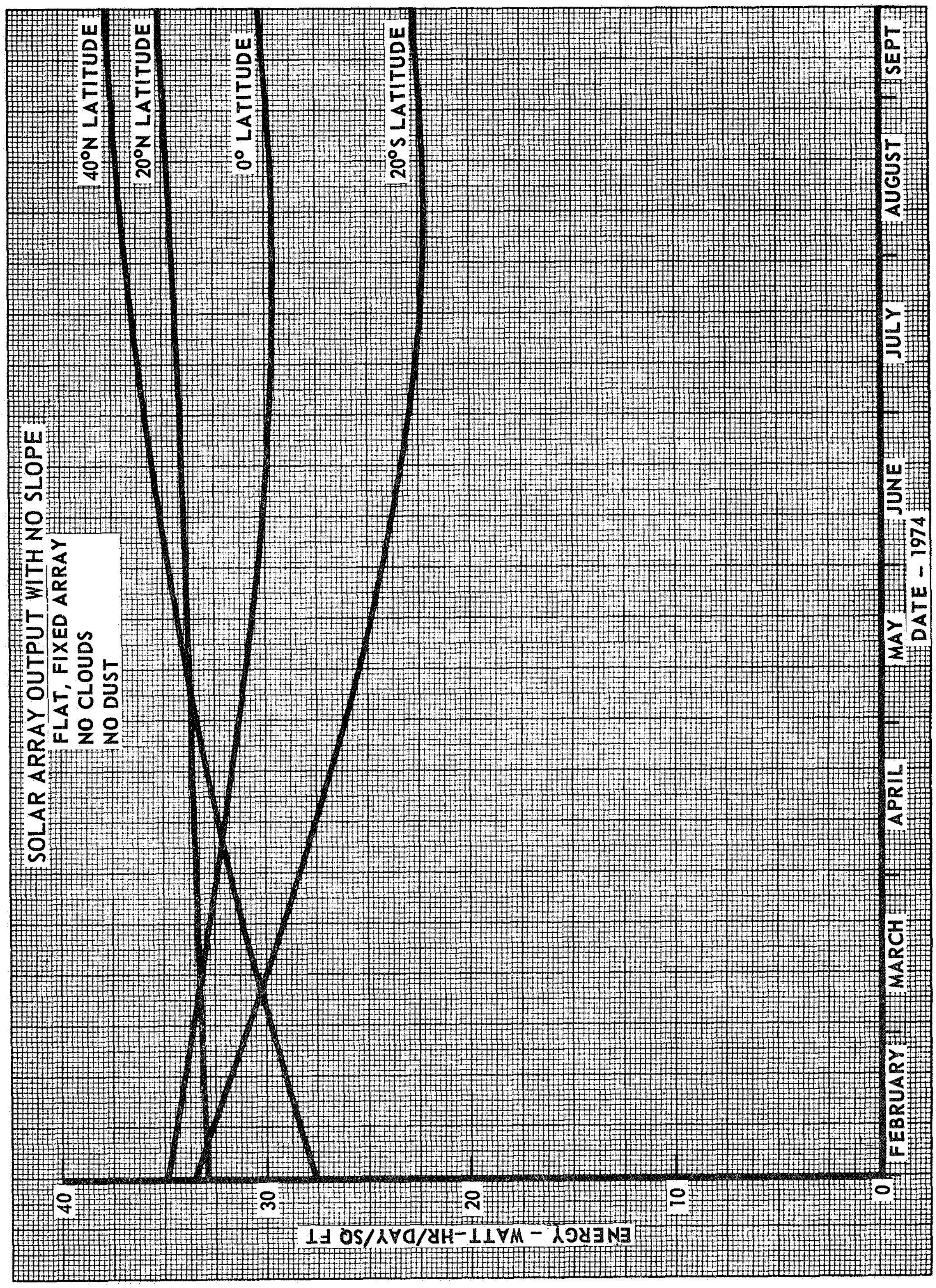

SOLAR ARRAY OUTPUT WITH NO SLOPE
FLAT, FIXED ARRAY
NO CLOUDS
NO DUST
40°N LATITUDE
20°N LATITUDE
0° LATITUDE
20°S LATITUDE
ENERGY - WATT-HR/DAY/SQ FT
0
10
20
30
40
FEBRUARY
MARCH
APRIL
MAY
JUNE
JULY
AUGUST
SEPT
DATE - 1974

FIGURE 1.4-3

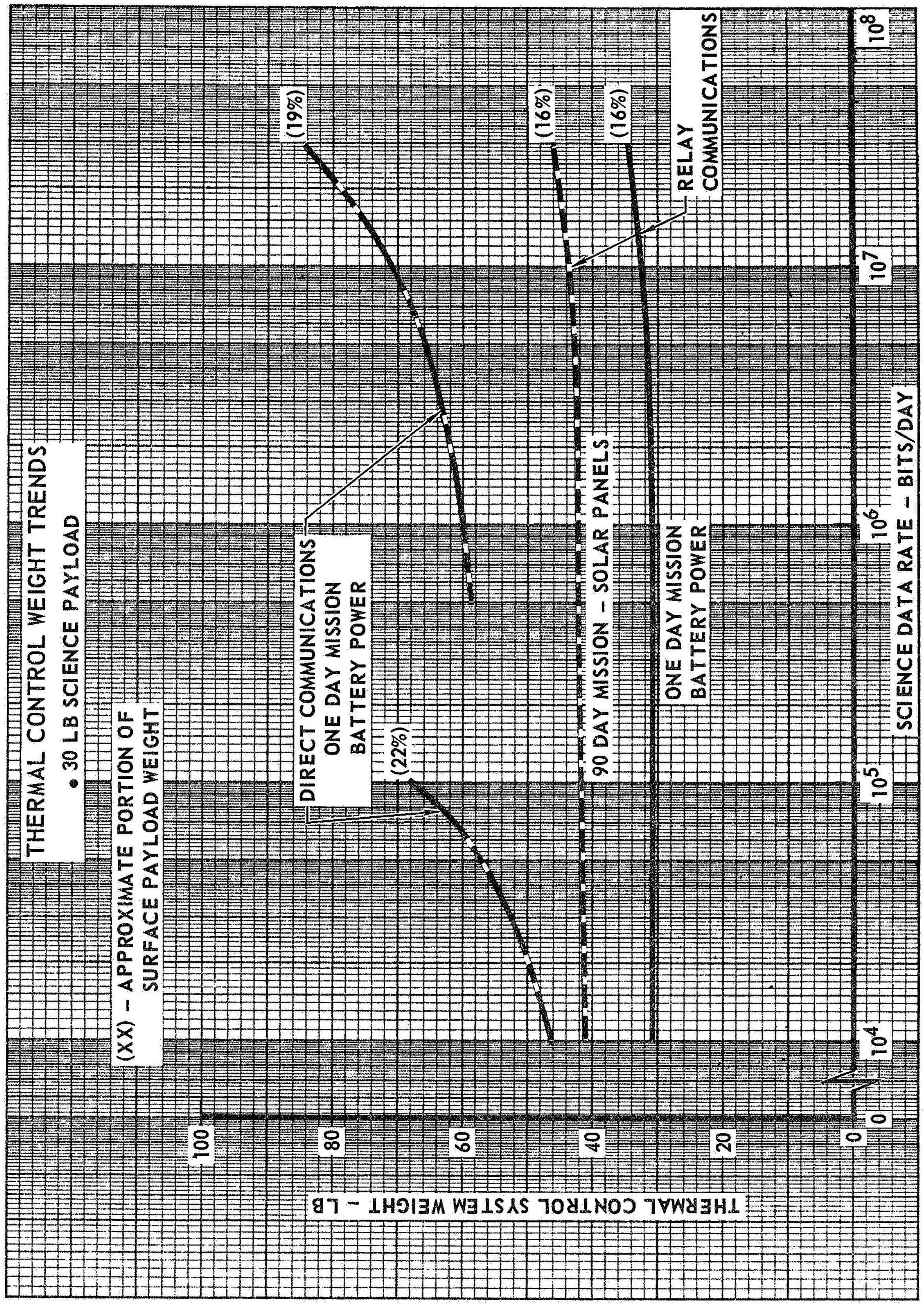
1.4.5 SEQUENCER - The sequencer system provides the means for accomplishing, independent of Earth command, the sequential time-based events for equipment checkout prior to deorbit, and for equipment control during separation, deorbit, entry, landing, and surface operations. Primary considerations in the analysis of the sequencer requirements include the memory type, possible integration of the pre-touchdown sequencing functions into the guidance computer, the pre-entry stabilization technique as it affects sequencer/computer integration, and the surface mission duration.

1.4.6 THERMAL CONTROL - During the flight phase the thermal control system uses electrical and/or isotope heaters, thermal coatings, and insulation to maintain the thermal environment in the capsule. Only when power is supplied by an RTG is there a significant heat rejection requirement during this period. As shown in Figure 1.4.-4 the portion of surface landed payload attributable to thermal control varies between 16 and 22%. Insulation necessary to protect equipment from the low temperature Martian night is the heaviest part of the lander thermal control. Other significant components include electrical and/or isotope heaters, and phase change material to decrease lander temperature sensitivity to environment fluctuations. Thermal control of the lander is complicated by the range of possible mission objectives, by the uncertainties in component performance after exposure to handling, sterilization, and mission environments, and by the uncertainty of the natural environments.

1.4.7 PROPULSION - The capsule requires propulsion to deorbit, to control attitude from capsule-orbiter separation to lander touchdown, and to effect the final deceleration required for a soft landing on the Mars surface.

For the deorbit functions, solid propellants provide a weight advantage over liquid monopropellant and bipropellant systems as shown in Figure 1.4-5. However, development of a sterilizable solid propellant rocket warrants immediate action to assure availability for the 1973 mission.

As shown in Figure 1.4-6, bipropellant terminal propulsion systems provide a performance advantage over monopropellant systems, but monopropellants offer an edge in simplicity, cost, and reliability which is particularly attractive for the 1973 mission.



THERMAL CONTROL WEIGHT TRENDS
● 30 LB SCIENCE PAYLOAD
(XX) - APPROXIMATE PORTION OF SURFACE PAYLOAD WEIGHT
DIRECT COMMUNICATIONS
ONE DAY MISSION
BATTERY POWER
(22%)
(19%)
(16%)
(16%)
90 DAY MISSION - SOLAR PANELS
ONE DAY MISSION
BATTERY POWER
RELAY
COMMUNICATIONS
THERMAL CONTROL SYSTEM WEIGHT - LB
100
80
60
40
20
0
0
10^4
10^5
10^6
10^7
10^8
SCIENCE DATA RATE - BITS/DAY

FIGURE 1.4-4

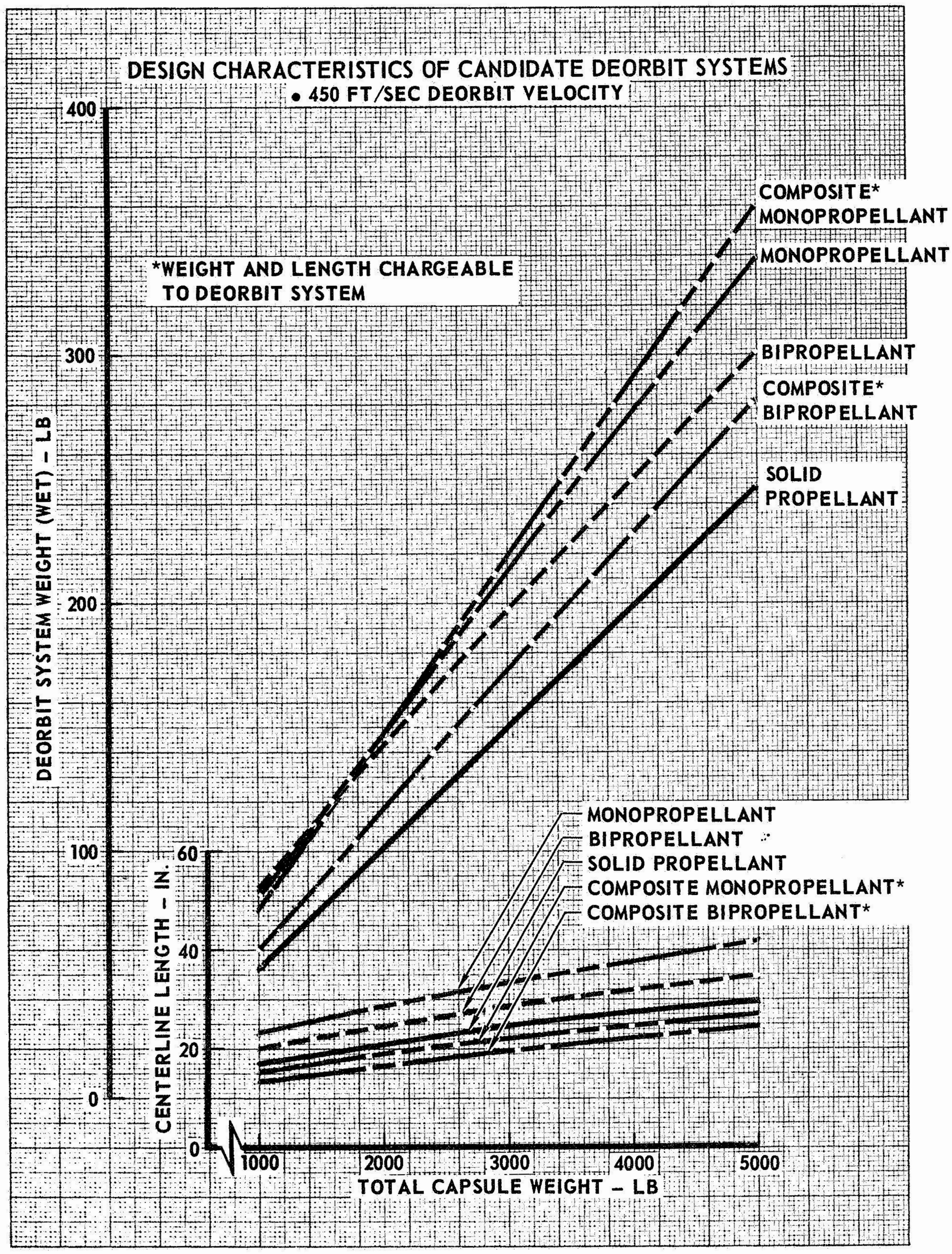
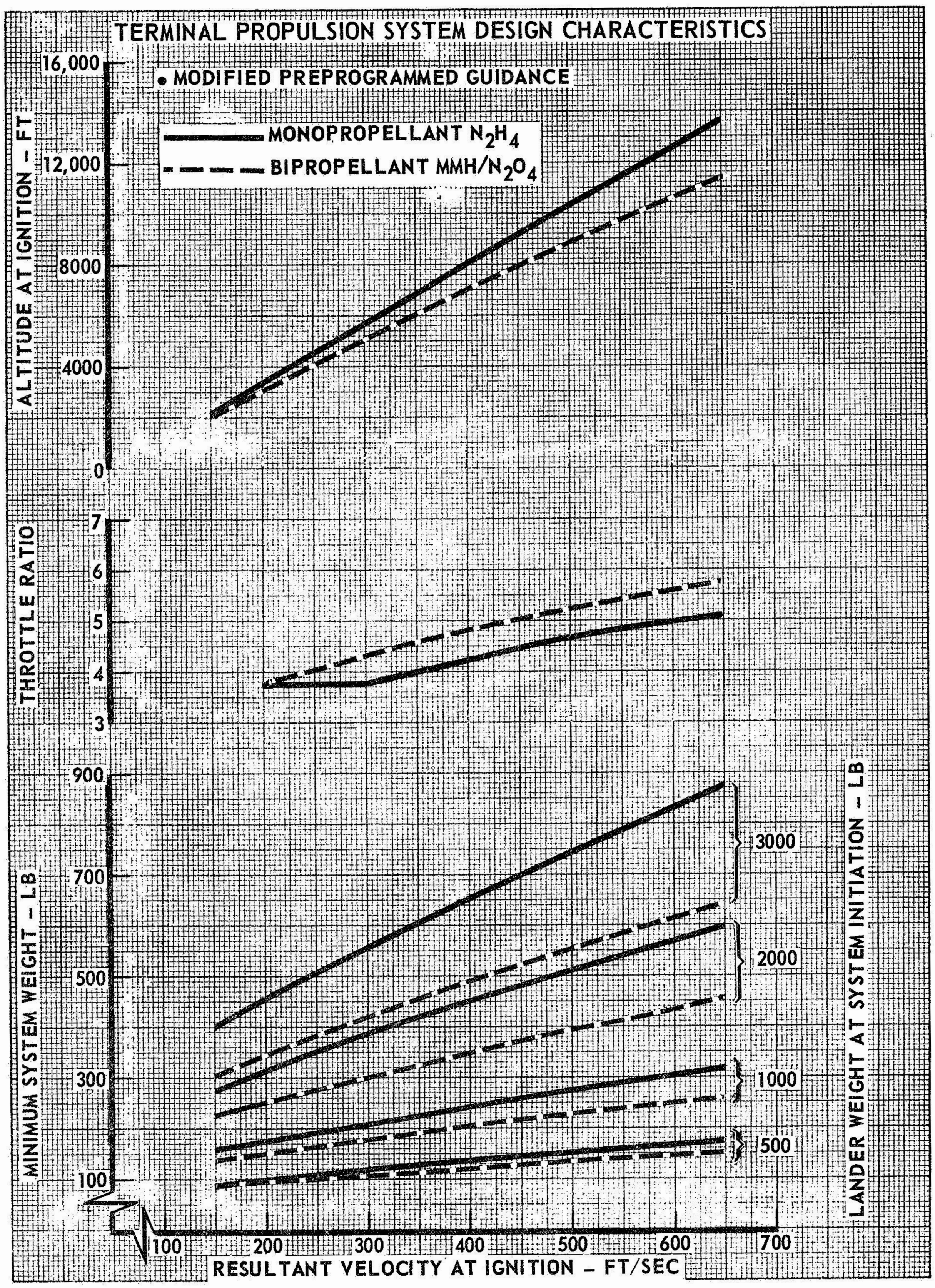

DESIGN CHARACTERISTICS OF CANDIDATE DEORBIT SYSTEMS
• 450 FT/SEC DEORBIT VELOCITY
*WEIGHT AND LENGTH CHARGEABLE TO DEORBIT SYSTEM
COMPOSITE* MONOPROPELLANT
MONOPROPELLANT
BIPROPELLANT
COMPOSITE* BIPROPELLANT
SOLID PROPELLANT
MONOPROPELLANT
BIPROPELLANT
SOLID PROPELLANT
COMPOSITE MONOPROPELLANT*
COMPOSITE BIPROPELLANT*
DEORBIT SYSTEM WEIGHT (WET) – LB
400
300
200
100
0
CENTERLINE LENGTH – IN.
60
40
20
0
1000
2000
3000
4000
5000
TOTAL CAPSULE WEIGHT – LB

FIGURE 1.4–5



TERMINAL PROPULSION SYSTEM DESIGN CHARACTERISTICS
• MODIFIED PREPROGRAMMED GUIDANCE
MONOPROPELLANT N_2H_4
BIPROPELLANT MMH/N_2O_4
ALTITUDE AT IGNITION – FT
16,000
12,000
8000
4000
0
THROTTLE RATIO
7
6
5
4
3
MINIMUM SYSTEM WEIGHT – LB
900
700
500
300
100
LANDER WEIGHT AT SYSTEM INITIATION – LB
3000
2000
1000
500
100
200
300
400
500
600
700
RESULTANT VELOCITY AT IGNITION – FT/SEC

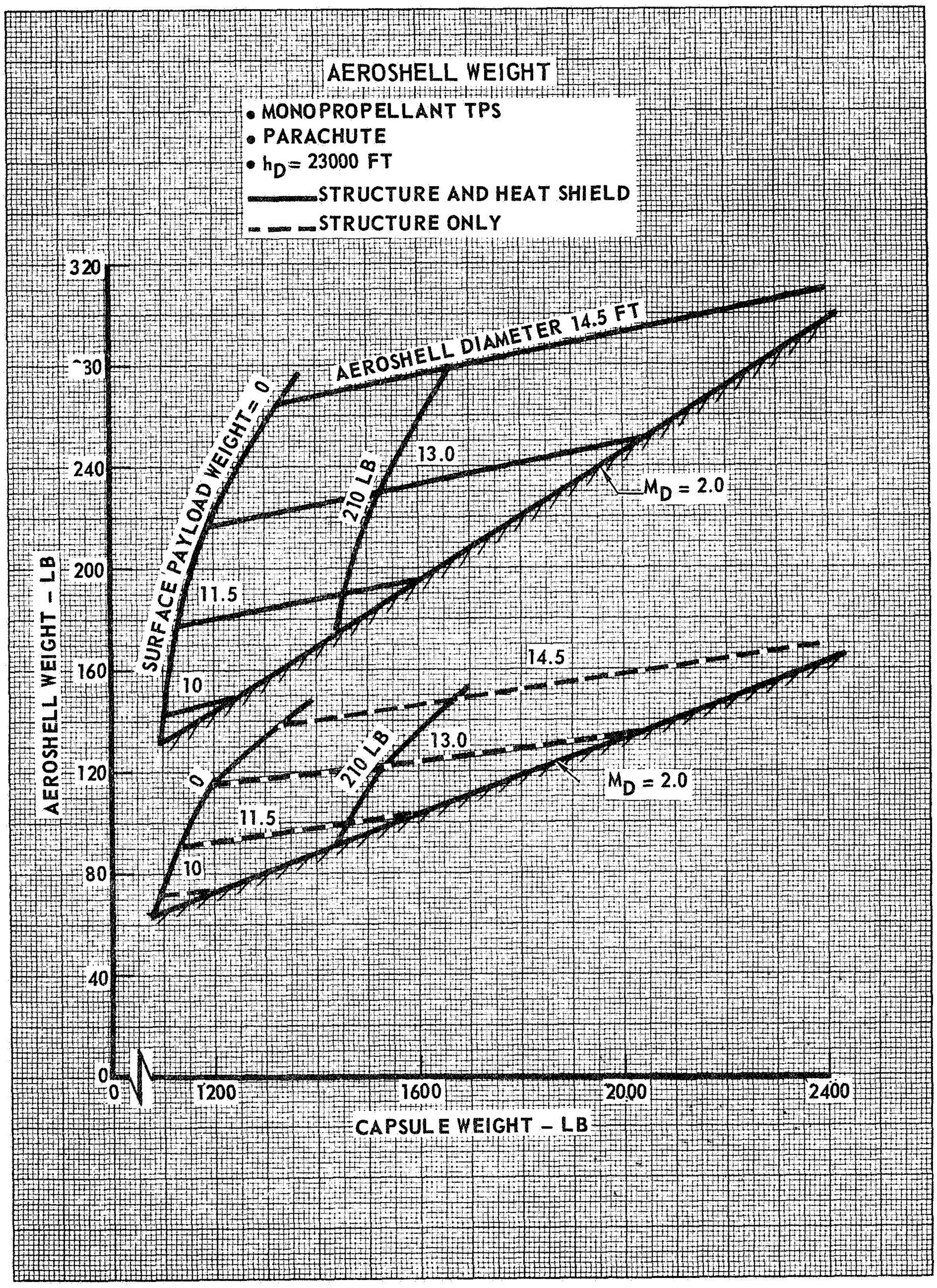
FIGURE 1.4-6

1.4.8 AUXILIARY AERODYNAMIC DECELERATOR - The auxiliary aerodynamic decelerator system decelerates the capsule to altitude and velocity conditions compatible with the initiation of the terminal propulsion system. In addition, the auxiliary aerodynamic decelerator provides stabilization and longer descent time; this permits acquisition of more and better quality atmospheric data. The characteristics of the entry capsules considered in this study ($M \leq 2$ at practical altitudes) permits the use of parachutes as the auxiliary aerodynamic decelerator device. For the present study, the NASA-Langley PEPP data is used extensively to define parachute configuration physical characteristics and performance parameters such as aerodynamic drag coefficient, stability, deployment-inflation times, and parachute opening shock loads.

Entry capsule concepts with high ballistic parameters which would otherwise exceed Mach 2 at usable altitudes may require auxiliary aerodynamic devices such as inflatable aeroshell extensions or attached inflatable decelerators.

1.4.9 AEROSHELL - The aeroshell decelerates the entry capsule and provides thermal protection to the capsule during ballistic entry into the Martian atmosphere. A 60° half-angle sphere-cone configuration, with a nose radius equal to one-half the base radius, was used for this study. Structure for the conical portion consists of a titanium single-faced, longitudinally corrugated shell with internal aluminum rings. Nose cap structure is of sandwich construction consisting of fiberglass face sheets and honeycomb core. Heat protection of the conical portion is provided by a low density charring ablator. Nose cap thermal protection is provided by non-ablative, hardened compacted fibers of alumino-silicate to prevent interference with atmospheric sampling experiments. A silica cloth curtain was used for thermal protection of the aeroshell base areas. Aeroshell weight and size are related as shown in Figure 1.4-7.

1.4.10 LANDING SYSTEM - The landing system limits the deceleration of the surface payload to less than 20 g's and assures that stability is maintained during landing. Stability is of primary concern because of the requirement for landing on ridges and cones and on $\pm 34°$ slopes. The Uni-Disc landing system used for this study consists of a base platform, crushable attenuation material, footpad, and tension cable assemblies. The surface payload, terminal descent and landing, are mounted on the base platform. The ratio of center-of-gravity



AEROSHELL WEIGHT
MONOPROPELLANT TPS
PARACHUTE
h_D = 23000 FT
STRUCTURE AND HEAT SHIELD
STRUCTURE ONLY
AEROSHELL DIAMETER 14.5 FT
SURFACE PAYLOAD WEIGHT = 0
210 LB
13.0
11.5
10
M_D = 2.0
14.5
0
13.0
210 LB
11.5
10
M_D = 2.0
AEROSHELL WEIGHT – LB
320
280
240
200
160
120
80
40
0
0
1200
1600
2000
2400
CAPSULE WEIGHT – LB

FIGURE 1.4–7

height to footpad diameter (H/R) is limited to .6 to provide landing stability on the 34° slope.

1.4.11 CANISTER AND ADAPTER - The canister maintains sterile conditions of the capsule, including lander and associated equipment, from terminal sterilization until capsule/orbiter separation in Mars orbit. Canister structure consists of forward and aft aluminum, semi-monocoque shell structures mechanically joined at the field joint ring. Canister separation occurs at this ring by a dual confined explosive separation device. A positive pressure differential is maintained in the canister during boost to prevent biological contamination. A venting system is installed in the canister to prevent excessive internal pressures.

The adapter is a simple truss assembly which joins the capsule to the aft canister. Canister inertia loads occurring during ground handling, launch, and cruise are transferred by the adapter structure to canister/adapter interface and then through the canister interface cone to the orbiter. Sixteen aluminum tubular members comprise the adapter structure used for this study.

1.5 Environment Sensitivity

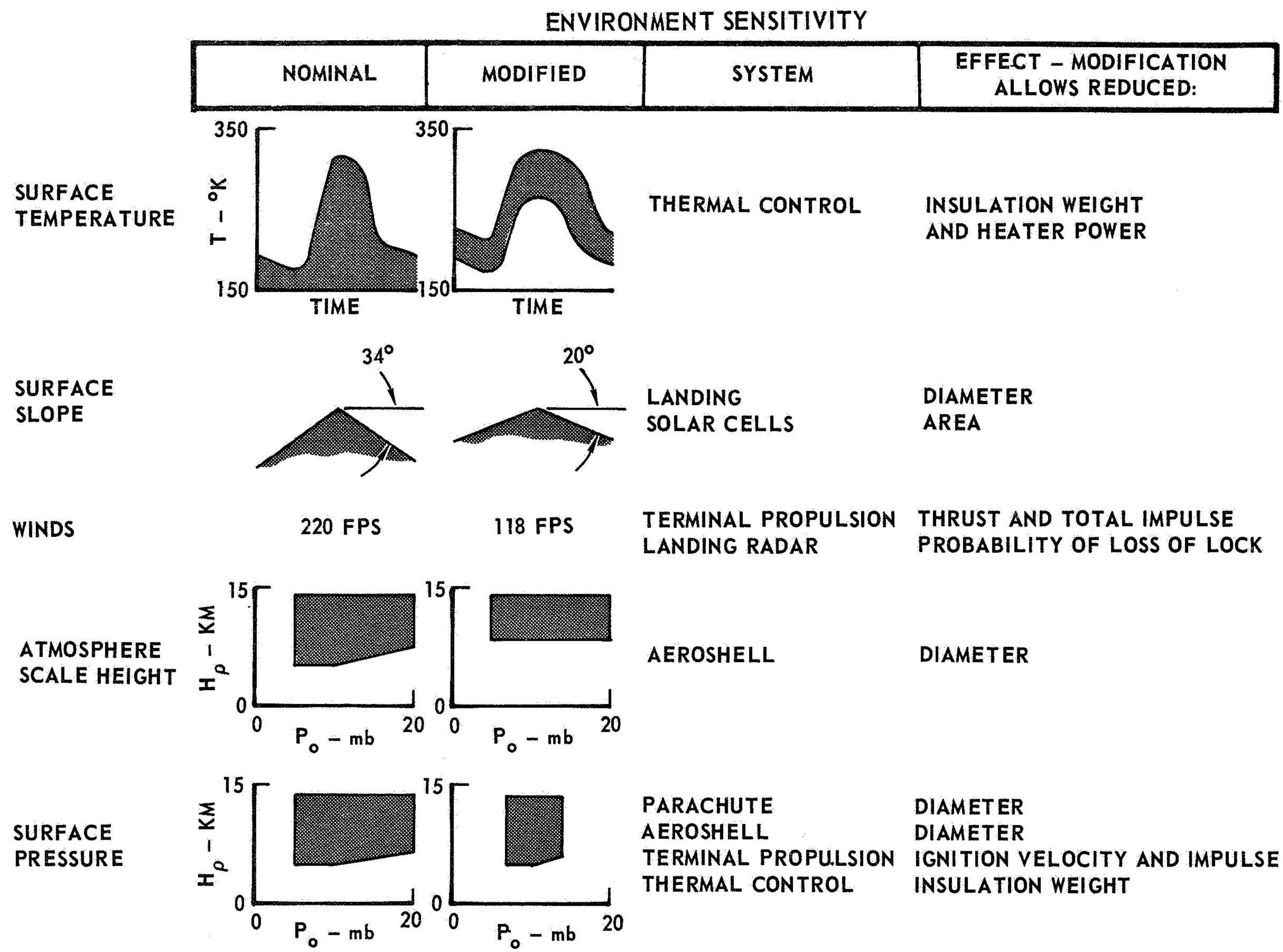
Evaluation of the effect of changes in the postulated environment included consideration of atmospheric scale height, surface pressure and temperature, winds, and surface slope. In addition to considering changes in each of these individually, a combination "modified environment" was studied, as shown in Figure 1.5-1.

The surface payload is especially sensitive to two environment factors: the surface slope and the temperature. For the minimum mission, in which capsule power is obtained from batteries, a surface payload weight reduction of more than 40 lbs can be made if the "cold day" specified in the VOYAGER Requirements and Constraints Document is removed as a design requirement for the low-latitude-landing soft landers. The effect on surface payload weight is illustrated in Figure 1.5-2. If electrical heaters are used this gain is amplified significantly as mission duration increases.

The surface slope affects the definition of the worst case for solar panel operation. Reduction to 20° slope as a design value would provide a minimum available energy value about 30% higher than with the 34° slope currently in use; see Figure 1.5-3.

The landing system is also affected by surface slope. In the range of lander sizes and weights studied, a nearly linear relationship was evidenced; see Figure 1.5-4. A change of surface slope design requirement changes lander system weight and therefore total capsule weight. For slopes less than 34°, a 6° increase in slope requires that an additional 1% of the landed weight be devoted to the system. Furthermore, since the Uni-Disc selection was based on its ability to handle the combination of high slopes and associated sharp peaks, a significant reduction in slope might reopen the selection of landing system type.

Low altitude winds affect the total velocity and flight path angle at terminal propulsion system ignition. In low density atmospheres a reduction from 220 ft/sec to 118 ft/sec will allow the terminal propulsion system to be sized for about 15% less velocity increment and a lower thrust level, saving



ENVIRONMENT SENSITIVITY
NOMINAL
MODIFIED
SYSTEM
EFFECT – MODIFICATION ALLOWS REDUCED:
SURFACE TEMPERATURE
T – °K
350
150
TIME
THERMAL CONTROL
INSULATION WEIGHT AND HEATER POWER
SURFACE SLOPE
34°
20°
LANDING
SOLAR CELLS
DIAMETER
AREA
WINDS
220 FPS
118 FPS
TERMINAL PROPULSION
LANDING RADAR
THRUST AND TOTAL IMPULSE
PROBABILITY OF LOSS OF LOCK
ATMOSPHERE SCALE HEIGHT
H_ρ – KM
15
0
20
P_o – mb
AEROSHELL
DIAMETER
SURFACE PRESSURE
PARACHUTE
AEROSHELL
TERMINAL PROPULSION
THERMAL CONTROL
DIAMETER
DIAMETER
IGNITION VELOCITY AND IMPULSE
INSULATION WEIGHT

FIGURE 1.5–1

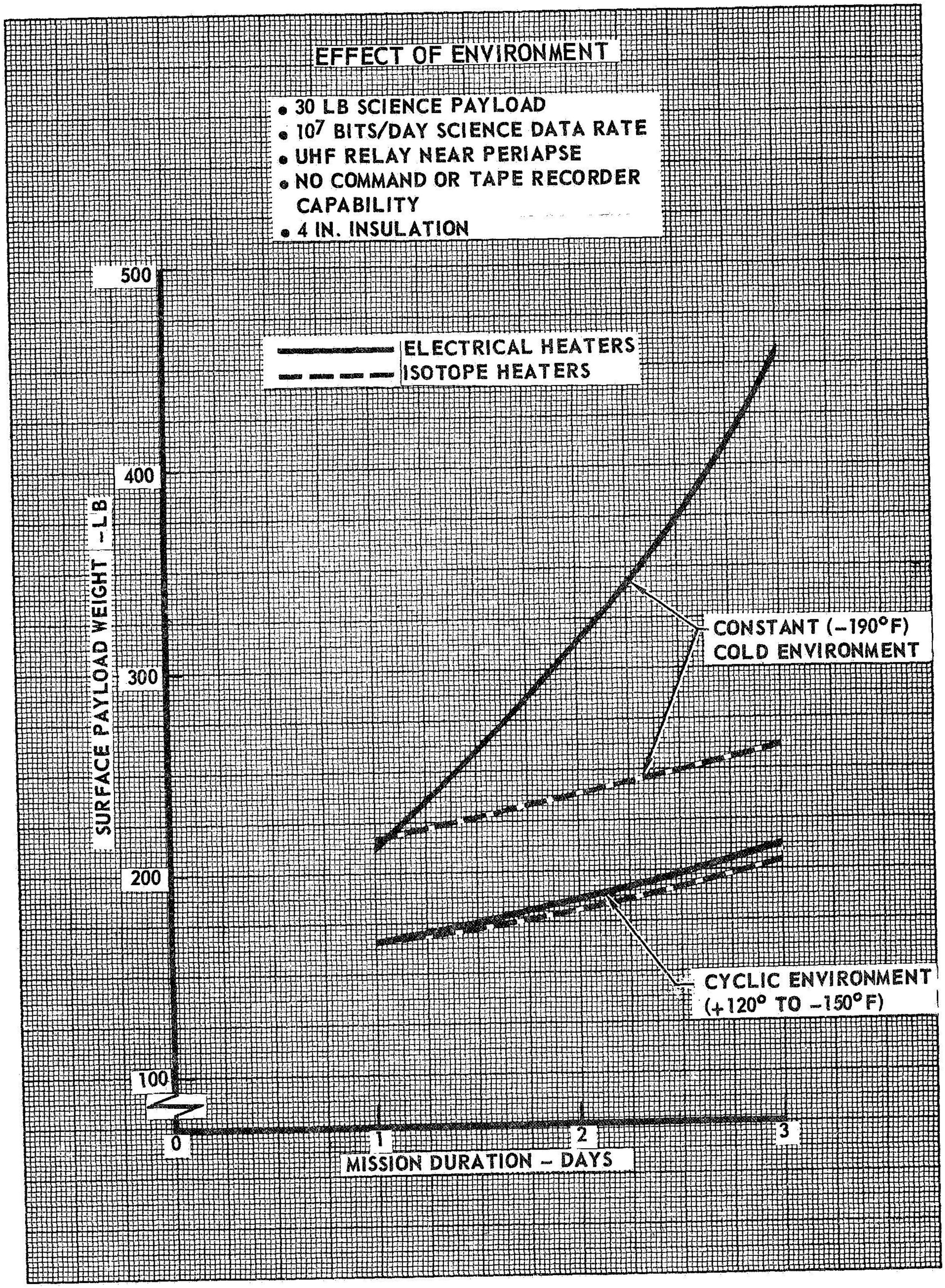

EFFECT OF ENVIRONMENT
• 30 LB SCIENCE PAYLOAD
• 10^7 BITS/DAY SCIENCE DATA RATE
• UHF RELAY NEAR PERIAPSE
• NO COMMAND OR TAPE RECORDER CAPABILITY
• 4 IN. INSULATION
ELECTRICAL HEATERS
ISOTOPE HEATERS
CONSTANT (–190°F) COLD ENVIRONMENT
CYCLIC ENVIRONMENT (+120° TO –150°F)
SURFACE PAYLOAD WEIGHT – LB
500
400
300
200
100
0
1
2
3
MISSION DURATION – DAYS

FIGURE 1.5–2

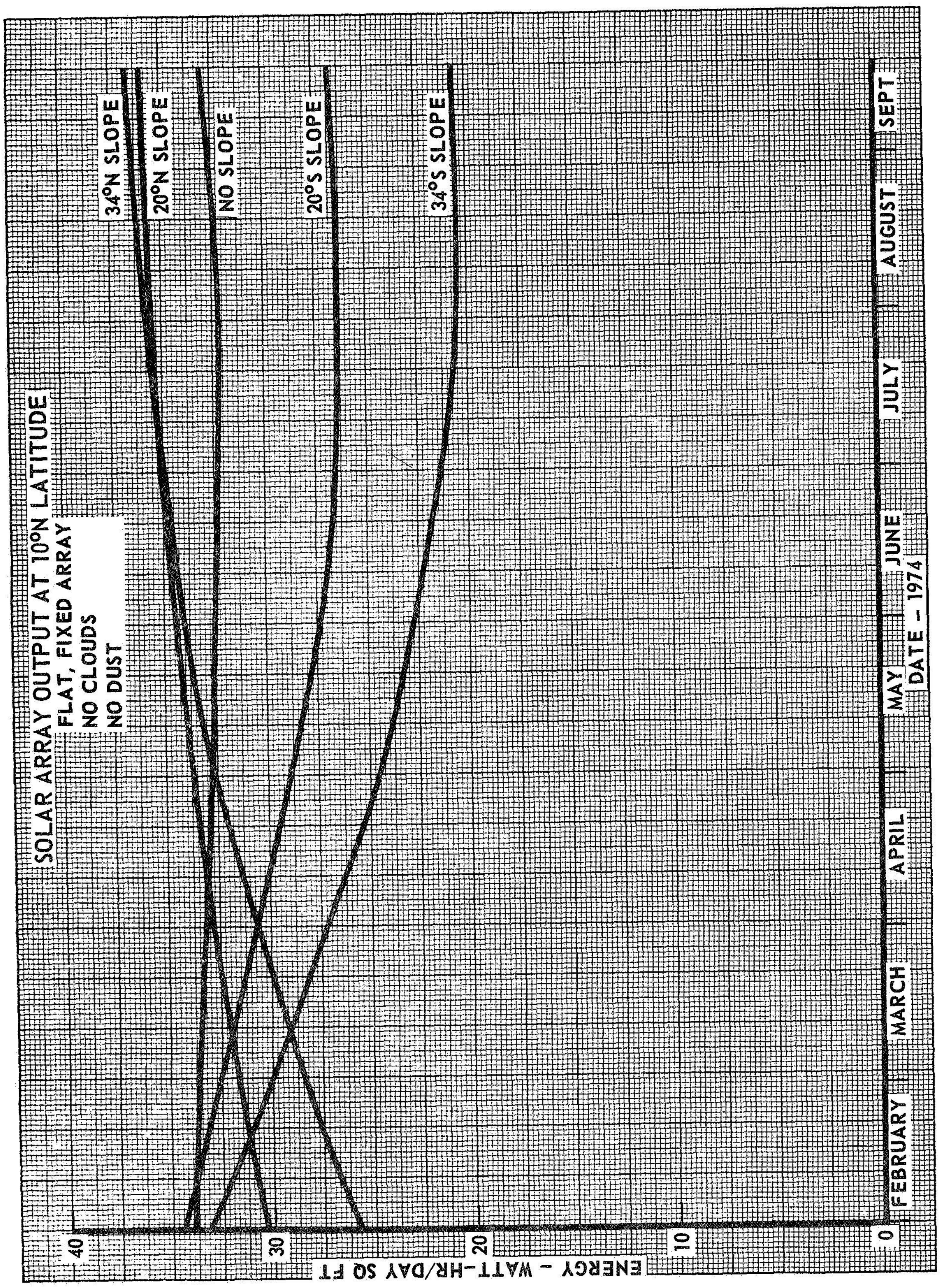

SOLAR ARRAY OUTPUT AT 10°N LATITUDE
FLAT, FIXED ARRAY
NO CLOUDS
NO DUST
34°N SLOPE
20°N SLOPE
NO SLOPE
20°S SLOPE
34°S SLOPE
40
30
20
10
0
ENERGY - WATT-HR/DAY SQ FT
FEBRUARY
MARCH
APRIL
MAY
JUNE
JULY
AUGUST
SEPT
DATE - 1974

FIGURE 1.5-3

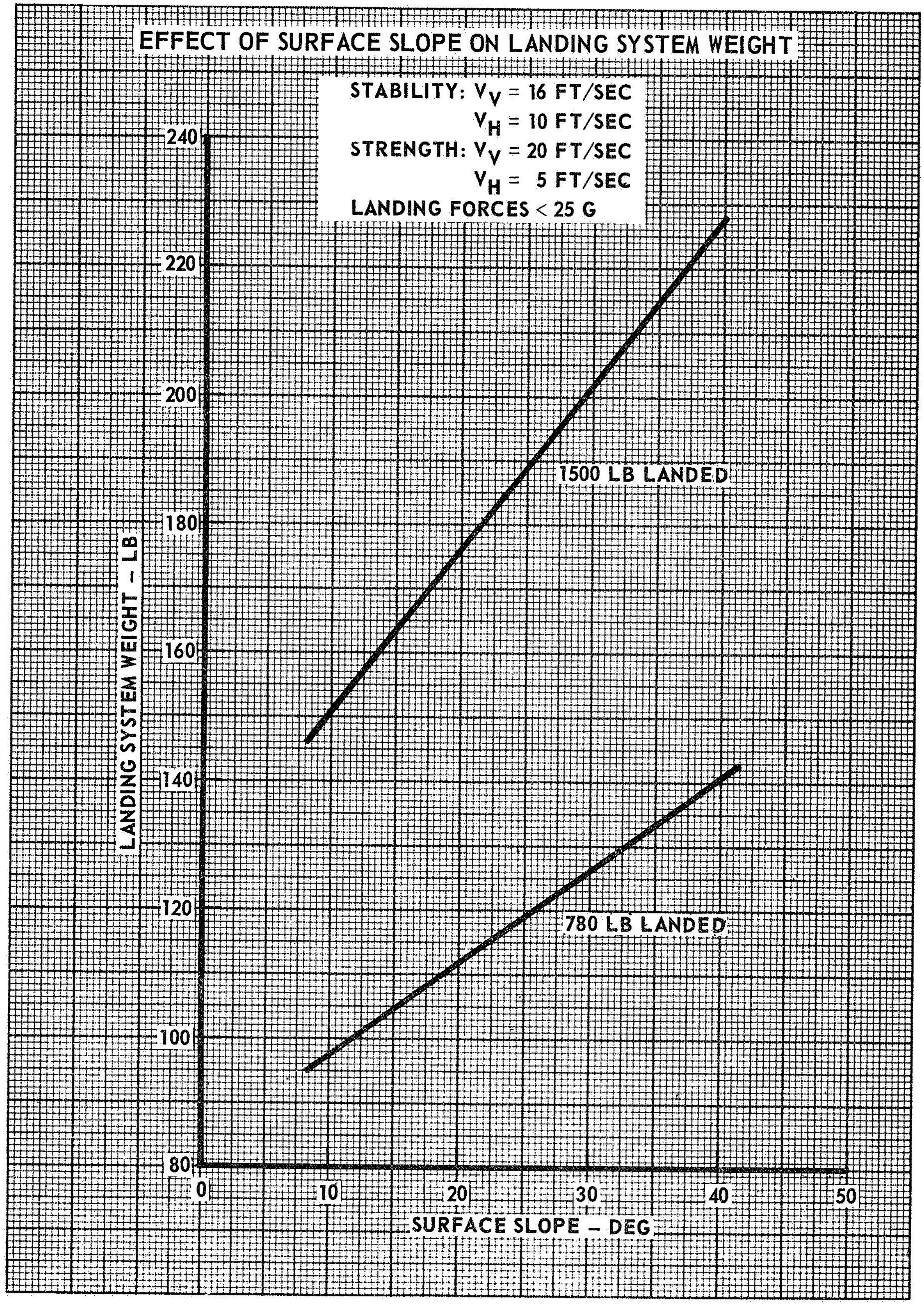

EFFECT OF SURFACE SLOPE ON LANDING SYSTEM WEIGHT
STABILITY: V_V = 16 FT/SEC
V_H = 10 FT/SEC
STRENGTH: V_V = 20 FT/SEC
V_H = 5 FT/SEC
LANDING FORCES < 25 G
240
220
200
180
160
140
120
100
80
LANDING SYSTEM WEIGHT – LB
1500 LB LANDED
780 LB LANDED
0
10
20
30
40
50
SURFACE SLOPE – DEG

FIGURE 1.5–4

both weight and cost. For the high density atmosphere, this reduction in wind would allow the flight path to be more nearly vertical and decrease the possibility of loss of radar lock during the attitude hold period and after velocity vector alignment.

Atmospheric pressure and density as a function of altitude have a significant effect on the capsule deceleration systems, specifically the aeroshell, parachute, and terminal propulsion system. The VM-1 through VM-10 atmospheres which were used as the basis for the study may be characterized by a surface pressure and a pressure or density scale height. The surface pressure of the VM atmospheres ranges from 5 to 20 millibars. Scale height, the altitude increment in which pressure or density changes by a factor of e (the base of natural logarithms) is constant above the tropopause; for the VM atmospheres it is between 5 and 14 kilometers. Surface pressure or density (for the VM atmospheres they are proportional for fixed scale height) is important primarily because it governs the parachute size required to achieve a given equilibrium vertical velocity. For a 300 ft/sec terminal vertical velocity - near optimum on a total capsule cost and weight basis - an increase from 5 to 7 mb surface pressure would allow parachute diameter to be decreased 14% if this were the only sizing criteria. Because aeroshell separation and descent time also influence parachute sizing, not all of this potential gain could actually be achieved. Surface atmospheric pressure influences engine design characteristics of the terminal propulsion system, but does not significantly affect the overall capsule design. However, an increase in surface pressure will reduce the optimum parachute phase terminal velocity and therefore change the sizing of the parachute and terminal propulsion systems.

In shallow atmospheres (low scale height), the capsule penetrates to low altitudes and high atmospheric density at higher speeds than in deep atmospheres. This causes higher dynamic pressure and heating rates, and more rapid deceleration, but for shorter periods, Therefore, for deceleration to a given speed at given altitude as required for deployment of an auxiliary aerodynamic decelerator-low scale height atmospheres require a lower ballistic parameter (m/C_DA), capability to withstand higher air loads, deceleration levels, and heating rates, but less heat protection equipment. For deceleration to Mach 2 at 23 000 feet, the nominal parachute deployment condition, an m/C_DA of not more than .25 slug/ft^2 is allowable for VM-2, -4, and -8, the 5 km scale height atmospheres, but a value of .33 slug/ft^2 is allowable in atmospheres with scale heights of 8 km or more, as shown in Figure 1.5-5.

High scale height atmospheres affect the VHF communications from capsule to orbiter by increasing the atmospheric descent time and thus reducing the period between landing and the time at which the orbiter passes beyond the horizon. This effect has not been examined in detail since reduction of maximum scale height seems less likely in the near future than increasing the minimum scale height and because its effect is primarily on operations and choice of trajectory for a particular mission rather than on capsule design.

A significantly more efficient capsule design results if the modifications noted in Figure 1.5-1 in temperature, surface slope, winds and atmospheric scale height are applied simultaneously. Figure 1.5-6 shows that a 15% reduction in weight can be achieved by this means for a minimum mission capsule and that this percentage increases as the capsule size increases.

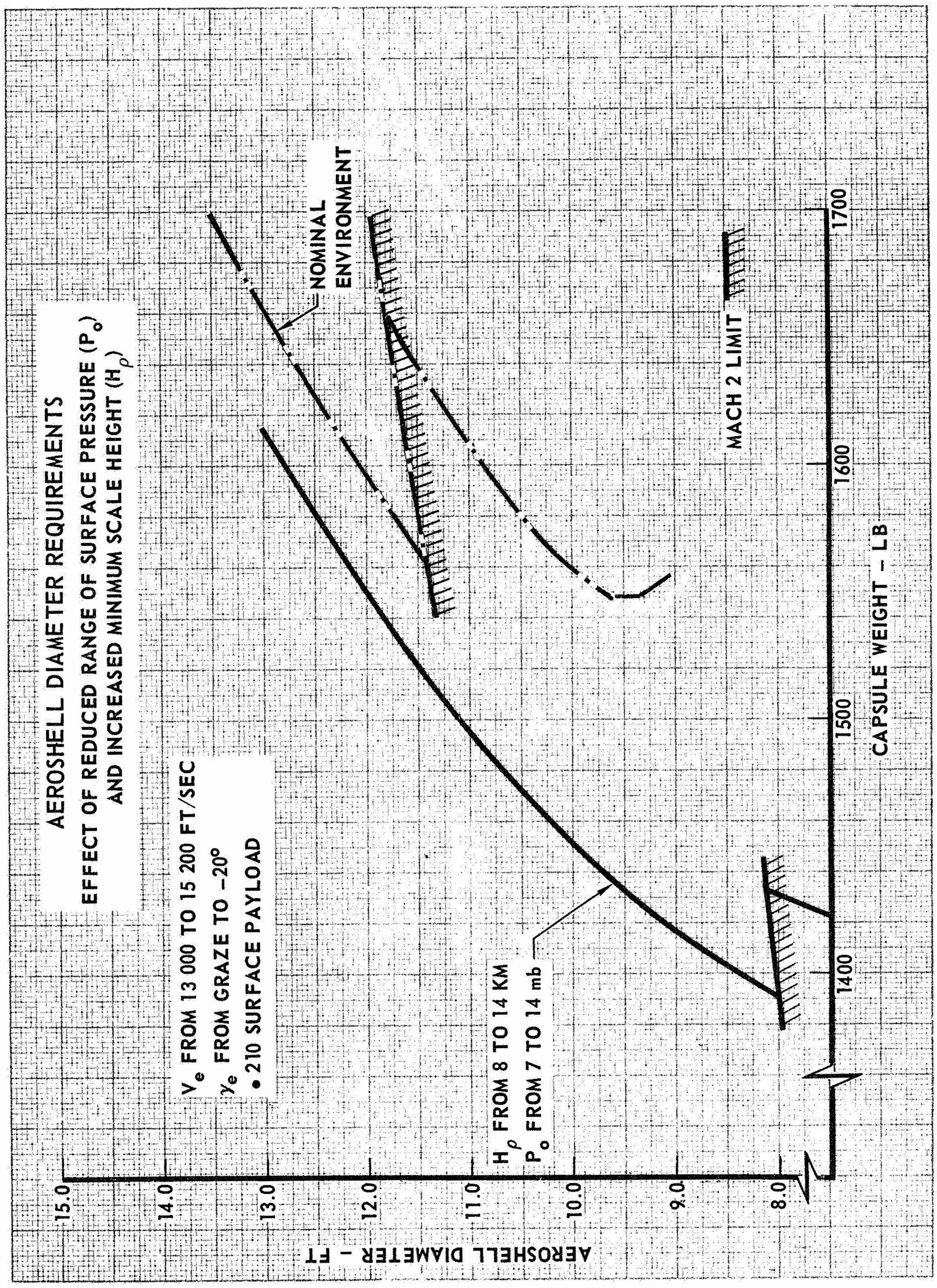

AEROSHELL DIAMETER REQUIREMENTS
EFFECT OF REDUCED RANGE OF SURFACE PRESSURE (P_o) AND INCREASED MINIMUM SCALE HEIGHT (H_ρ)
V_e FROM 13 000 TO 15 200 FT/SEC
γ_e FROM GRAZE TO −20°
• 210 SURFACE PAYLOAD
H_ρ FROM 8 TO 14 KM
P_o FROM 7 TO 14 mb
NOMINAL ENVIRONMENT
MACH 2 LIMIT
AEROSHELL DIAMETER – FT
15.0
14.0
13.0
12.0
11.0
10.0
9.0
8.0
1400
1500
1600
1700
CAPSULE WEIGHT – LB

FIGURE 1.5–5

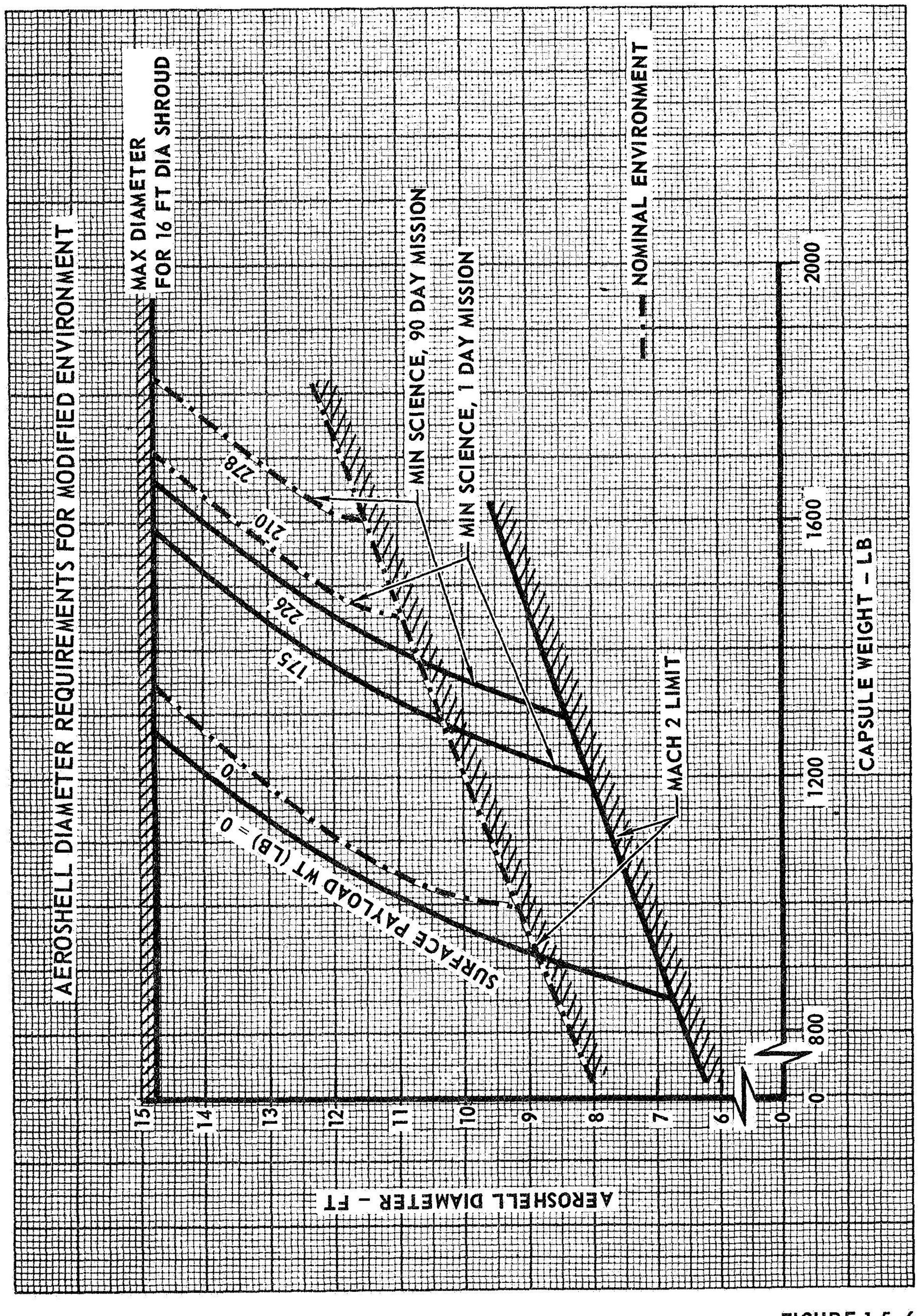

AEROSHELL DIAMETER REQUIREMENTS FOR MODIFIED ENVIRONMENT
MAX DIAMETER
FOR 16 FT DIA SHROUD
MIN SCIENCE, 90 DAY MISSION
MIN SCIENCE, 1 DAY MISSION
NOMINAL ENVIRONMENT
MACH 2 LIMIT
SURFACE PAYLOAD WT (LB) = 0
0
175
225
210
278
AEROSHELL DIAMETER – FT
15
14
13
12
11
10
9
8
7
6
0
CAPSULE WEIGHT – LB
0
800
1200
1600
2000

FIGURE 1.5–6

1.6 Conceptual Designs

Five design concepts were initially selected to provide insight into the capsule integration and to provide confirmation of the parametric analyses. Consideration of Concept IV, selected to provide near maximum capability within the constraints, was terminated midway in the study to allow concentration on the smaller capsules more suitable for the 1973 missions. The remaining four exhibit the effects of variations in lifetime and design criteria on capsules with the minimum, 30 lb, surface science payload. Concepts I and II were designed for minimum mission duration - at least one complete diurnal cycle. Concepts III and V provide surface mission duration of 90 days or more; use of solar cells to recharge batteries permits this extension. Concepts II and III satisfy the full range of nominal environment specifications and the mission flexibility criteria listed in Table 1-1. Concepts I and V were designed to satisfy less rigorous environment and flexibility requirements. These two concepts have the advantage of compatibility with a 10 ft diameter booster payload shroud, thus obviating hammerheading, as well as reduced cost for the capsule itself.

The capsule mission for each of these concepts includes a boost and Earth orbit phase, an interplanetary phase, and a Mars orbit phase. The capsule then separates from the orbiter, performs a deorbit maneuver, descends, and enters the atmosphere. Trajectory parameters for a typical mission are presented in Table 1.6-1.

Characteristics of the five concepts are presented in Table 1.6-2. Table 1.6-3 describes the capsule systems for Concepts I, II, III, and V. Each of these concepts utilizes a parachute and a three engine monopropellant terminal propulsion system. The parachute was selected despite a small penalty in weight and an increase of about two feet in aeroshell diameter because it offered less development risk, an important factor for the 1973 mission. The monopropellant system, though it leads to a heavier capsule than would a bi-propellant system, offers substantial savings because of a less costly development program. No difficulty is anticipated in the development of the 400 to 600 lb thrust monopropellants required for these capsules. Solid propellant rockets were selected for deorbit in these concepts because of their weight

TABLE 1.6-1

NOMINAL TRAJECTORY

• LAUNCH	
DATE	19 SEPT 1973 (TYPE I)
VIS VIVA ENERGY, C_3	44.06 KM^2/SEC^2
• ARRIVAL	
DATE	17 MAY 1974
FLIGHT TIME, T_F	240 DAYS
HYPERBOLIC EXCESS VELOCITY, V_{HP}	2.86 KM/SEC
• ORBIT (POSIGRADE NORTH)	
SIZE	SYNCHRONOUS (1000–33 124 KM)
INCLINATION, i	60°
ASCENDING NODE, Ω	291.3° (EQUATORIAL, MARS EQUINOX)
ARGUMENT OF PERIAPSE, ω	114.5
INSERTION VELOCITY INCREMENT, ΔV	2.1 KM/SEC
PERIAPSE ROTATION ANGLE, ρ	66°
• DEORBIT	
TRUE ANOMALY, θ_{DO}	214°
VELOCITY INCREMENT, ΔV	361.3 FT/SEC
DEFLECTION ANGLE, δ	+ 25.7°
DESCENT TIME	3.4 HRS
• ENTRY (800 000 FT ALTITUDE)	
VELOCITY, V_e	15 110 FT/SEC
FLIGHT PATH ANGLE, γ_e	–16°
TRUE ANOMALY, θ_e	331.6
CAPSULE LEAD ANGLE, $\Delta\theta_e$	5.35°
• TERMINAL DECELERATION	
PARACHUTE DEPLOYMENT ALTITUDE	23 000 FT
ALTITUDE AEROSHELL IMPACT (VM-7)	5000 TO 6000 FT
TERMINAL PROPULSION INITIATION	6500 FT
• LANDING	
SOLAR ANGLE, Γ_S	60°
LATITUDE, δ	10°N
CENTRAL ANGLE FROM ENTRY	16.8°
TIME FROM ENTRY	440 SEC
LANDING TRUE ANOMALY	348°
ORBITER TRUE ANOMALY	348°
POST-LANDED VIEW TIME (20° GROUND SLOPE)	435 SEC

TABLE 1.6-2

CONCEPTUAL DESIGN CHARACTERISTICS SUMMARY

	I	II	III	IV*	V
ENTRY SCIENCE	ATMOSPHERICS	ATMOSPHERICS	ATMOSPHERICS	ATMOSPHERICS DESCENT VIDEO	ATMOSPHERICS
SURFACE SCIENCE	IMAGING ATMOSPHERICS SOIL COMPOSITION	IMAGING ATMOSPHERICS SOIL COMPOSITION	IMAGING ATMOSPHERICS SOIL COMPOSITION	IMAGING ATMOSPHERICS SOIL COMPOSITION SUBSURFACE PROBE	IMAGING ATMOSPHERICS SOIL COMPOSITION
LIFETIME	1 DAY	1 DAY	90 DAYS	2 YEARS	90 DAYS
POWER	BATTERY	BATTERY	SOLAR PANELS	RTG	SOLAR PANELS
THERMAL CONTROL	ISOTOPE HEATERS	ELECTRIC HEATERS	ELECTRIC + ISOTOPE HEATERS		ISOTOPE HEATERS
LANDED COMMUNICATIONS	RELAY	RELAY	RELAY + LOW RATE DIRECT S-BAND	DIRECT	RELAY + LOW RATE DIRECT S-BAND
TERMINAL PROPULSION	MONOPROPELLANT	MONOPROPELLANT	MONOPROPELLANT	BIPROPELLANT	MONOPROPELLANT
DECELERATOR	PARACHUTE	PARACHUTE	PARACHUTE	AIDS	PARACHUTE
DEORBIT PROPULSION	SOLID PROPELLANT	SOLID PROPELLANT	SOLID PROPELLANT	SOLID PROPELLANT	SOLID PROPELLANT
ENTRY VELOCITY	14,000 TO 15,200	13,000 TO 15,200	13,000 TO 15,200	13,000 TO 15,200	14,000 TO 15,200
ENTRY ANGLE	−15° TO −18°	GRAZE TO −20°	GRAZE TO −20°	GRAZE TO −20°	−15° TO −18°
ATMOSPHERES	H_ρ 8 KM (MIN)	VM-1 TO VM-10	VM-1 TO VM-10	VM-1 TO VM-10	H_ρ = 8 KM (MIN)
SURFACE TEMP	CYCLIC	CYCLIC OR CONT COLD	CYCLIC + ONE CONT COLD		CYCLIC
SURFACE SLOPES	< 20°	< 34°	< 34°		< 20°
SURFACE PAYLOAD WEIGHT	176 LB	210 LB	278 LB	1060 LB	226 LB
CAPSULE WEIGHT	1226 LB	1450 LB	1603 LB	3500 LB	1313 LB
AEROSHELL DIA	8.83 FT	10.9 FT	11.5 FT	14.5 FT	8.83 FT

*EFFORT DISCONTINUED AT NASA – LRC REQUEST TO CONCENTRATE ON CAPSULES FOR 1973 MISSION

TABLE 1.6-3
CONCEPTUAL DESIGNS – SYSTEM CHARACTERISTICS SUMMARY

SYSTEM	CONCEPT I	CONCEPT II	CONCEPT III	CONCEPT V
SCIENCE				
ENTRY				
MASS SPECTROMETER				
WEIGHT	9 LB	9 LB	9 LB	9 LB
POWER	8 W	8 W	8 W	8 W
DATA RATE	80 BPS	80 BPS	80 BPS	80 BPS
ACCELEROMETER				
WEIGHT	2 LB	2 LB	2 LB	2 LB
POWER	4 W	4 W	4 W	4 W
DATA RATE	150 BPS	150 BPS	150 BPS	150 BPS
PRESSURE SENSORS				
WEIGHT	5 LB	5 LB	5 LB	5 LB
POWER	3 W	3 W	3 W	3 W
DATA RATE	48 BPS	48 BPS	48 BPS	48 BPS
TEMPERATURE SENSORS				
WEIGHT	1 LB	1 LB	1 LB	1 LB
POWER	0.02 W	0.02 W	0.02 W	0.02 W
DATA RATE	16 BPS	16 BPS	16 BPS	16 BPS
HUMIDITY				
WEIGHT	1 LB	1 LB	1 LB	1 LB
POWER	0.02 W	0.02 W	0.02 W	0.02 W
DATA RATE	8 BPS	8 BPS	8 BPS	8 BPS
LANDED				
CAMERAS				
WEIGHT	10 LB	10 LB	10 LB	10 LB
POWER	15 W	15 W	15 W	15 W
DATA RATE	10^7 BPD	10^7 BPD	10^7 BPD *	10^7 BPD *
ATMOSPHERIC PACKAGE				
WEIGHT	10 LB	10 LB	10 LB	10 LB
POWER	7 W	7 W	7 W	7 W
DATA RATE–RELAY	9.4×10^3 BPD	9.4×10^3 BPD	9.4×10^3 BPD	9.4×10^3 BPD
DATA RATE–DIRECT	–	–	10^3 BPD	10^3 BPD
ALPHA SPECTROMETER				
WEIGHT	10 LB	10 LB	10 LB	10 LB
POWER	2 W	2 W	2 W	2 W
DATA RATE	8.4×10^3 BPD	8.4×10^3 BPD	8.4×10^3 BPD	8.4×10^3 BPD
COMMUNICATIONS				
ENTRY TELEMETRY				
FREQUENCY	400 MHz	400 MHz	400 MHz	400 MHz
TRANSMITTER	3 W	3 W	3 W	3 W
TELEMETRY CHANNELS	401	401	401	401
DATA RATES	440 & 1320 BPS	440 & 1320 BPS	440 & 1320 BPS	440 & 1320 BPS
STORAGE TIME	120 & 60 SEC	150 & 50 SEC	150 & 50 SEC	120 & 60 SEC
STORAGE BITS	158,720	198,656	198,656	158,720
RECEIVING ANTENNA BW	120°	120°	120°	120°
TRANSMITTING ANTENNA BW	90°	90°	90°	90°
MODULATION	2 FSK	2 FSK	2 FSK	2 FSK
LANDED RELAY TELEMETRY				
FREQUENCY	400 MHz	400 MHz	400 MHz	400 MHz
TRANSMITTER	5 W	5 W	5 W	5 W
TELEMETRY CHANNELS	116	116	116	116
DATA RATE	41.6 KBPS	41.6 KBPS	41.6 KBPS	41.6 KBPS
STORAGE TIME	24.6 HR	24.6 HR	24.6 HR	24.6 HR
STORAGE BITS	167,936	167,936	167,936	167,936
RECEIVING ANTENNA BW (ORBITER)	120°	120°	120°	120°
TRANSMITTING ANTENNA BW	120°	120°	120°	120°
MODULATION	2 FSK	2 FSK	2 FSK	2 FSK
LANDED DIRECT TELEMETRY				
FREQUENCY			2300 MHz	2300 MHz
TRANSMITTER			20 W	20 W
TELEMETRY CHANNELS			70	70
DATA RATE			1.088 BPS	1.088 BPS
STORAGE TIME			24.6 HR	24.6 HR
STORAGE BITS			8192	8192
RECEIVING ANTENNA BW			0.15°	0.15°
TRANSMITTING ANTENNA BW			120°	120°
MODULATION			MFSK	MFSK

*0 BPD AFTER ORBITER IS DE-SYNCHRONIZED (FOR OTHER MISSIONS).

TABLE 1.6-3
CONCEPTUAL DESIGNS – SYSTEM CHARACTERISTICS SUMMARY (Continued)

SYSTEM	CONCEPT I	CONCEPT II	CONCEPT III	CONCEPT V
COMMUNICATIONS (Cont.)				
LANDED DIRECT COMMAND				
FREQUENCY			2300 MHz	2300 MHz
TRANSMITTER			100 KW	100 KW
COMMAND RATE			1 BPS	1 BPS
RECEIVING ANTENNA BW			120°	120°
TRANSMITTING ANTENNA BW			0.15°	0.15°
MODULATION			PSK-BCH	PSK-BCH
POWER				
LANDER:				
BATTERY –				
TYPE	Ag-Zn	Ag-Zn	Ag-Zn	Ag-Zn
CELL SIZE	15 A-HR	15 A-HR	15 A-HR	15 A-HR
CAPACITY	425 W-HR	425 W-HR	425 W-HR	425 W-HR
WEIGHT	22 LB	22 LB	22 LB	22 LB
SURFACE PAYLOAD:				
BATTERY –				
TYPE	Ag-Zn	Ag-Zn	Ag-Zn	Ag-Zn
CELL SIZE	12 A-HR	80 A-HR	45 A-HR	12.5 A-HR
CAPACITY	330 W-HR	2210 W-HR	1245 W-HR	350 W-HR
WEIGHT	14 LB	61 LB	37 LB	15 LB
DEPTH OF DISCHARGE	100%	100%	12%	45%
SOLAR ARRAY –				
CONFIGURATION	–	–	FLAT, FIXED	FLAT, FIXED
AREA	–	–	20 SQ FT	16 SQ FT
MINIMUM OUTPUT	–	–	410 W-HR/DAY	416 W-HR/DAY
MAXIMUM OUTPUT	–	–	733 W-HR/DAY	580 W-HR/DAY
WEIGHT	–	–	30 LB	24 LB
GUIDANCE & CONTROL				
CONTROL ELECTRONICS				
INERTIAL MEASUREMENT UNIT				
GYROS				
TYPE	SINGLE DEGREE-OF-FREEDOM, RATE INTEGRATING; PULSE-ON-DEMAND REBALANCE ELECTRONICS	SINGLE DEGREE-OF-FREEDOM, RATE INTEGRATING; PULSE-ON-DEMAND REBALANCE ELECTRONICS	SINGLE DEGREE-OF-FREEDOM, RATE INTEGRATING; PULSE-ON-DEMAND REBALANCE ELECTRONICS	SINGLE DEGREE-OF-FREEDOM, RATE INTEGRATING; PULSE-ON-DEMAND REBALANCE ELECTRONICS
g-INSENSITIVE DRIFT	±1.0 DEG/HR (3σ)	±1.0 DEG/HR (3σ)	±1.0 DEG/HR (3σ)	±1.0 DEG/HR (3σ)
MASS UNBALANCE DRIFTS	±1.5 DEG/HR/g (3σ)	±1.5 DEG/HR/g (3σ)	±1.5 DEG/HR/g (3σ)	±1.5 DEG/HR/g (3σ)
TORQUEING RATE	±60 DEG/SEC MAX	±60 DEG/SEC MAX	±60 DEG/SEC MAX	±60 DEG/SEC MAX
ACCELEROMETER TYPE	PENDULOUS FORCE REBALANCE; PULSE-ON-DEMAND REBALANCE ELECTRONICS	PENDULOUS FORCE REBALANCE; PULSE-ON-DEMAND REBALANCE ELECTRONICS	PENDULOUS FORCE REBALANCE; PULSE-ON-DEMAND REBALANCE ELECTRONICS	PENDULOUS FORCE REBALANCE; PULSE-ON-DEMAND REBALANCE ELECTRONICS
BIAS	$\pm 150 \times 10^{-6}$ g (3σ)	$\pm 150 \times 10^{-6}$ g (3σ)	$\pm 150 \times 10^{-6}$ g (3σ)	$\pm 150 \times 10^{-6}$ g (3σ)
SCALE FACTOR STABILITY	±0.015% (3σ)	±0.015% (3σ)	±0.015% (3σ)	±0.015% (3σ)
RANGE	0 TO 30 g	0 TO 30 g	0 TO 30 g	0 TO 30 g
CENTRAL COMPUTER				
TYPE	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT, AND POST-SEPARATION TO LANDING SEQUENCING	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT, AND POST-SEPARATION TO LANDING SEQUENCING	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT, AND POST-SEPARATION TO LANDING SEQUENCING	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT, AND POST-SEPARATION TO LANDING SEQUENCING
MEMORY LOGIC				
TYPE	MAGNETIC CORE	MAGNETIC CORE	MAGNETIC CORE	MAGNETIC CORE
ARITHMETIC	SERIAL	SERIAL	SERIAL	SERIAL
CAPACITY	4096 WORDS 20 BIT DATA WORD 512 WORDS UPDATEABLE	4096 WORDS 20 BIT DATA WORD 512 WORDS UPDATEABLE	4096 WORDS 20 BIT DATA WORD 512 WORDS UPDATEABLE	4096 WORDS 20 BIT DATA WORD 512 WORDS UPDATEABLE
SPEED	14 μSEC ADD TIME 154 μSEC MULTIPLY TIME	14 μSEC ADD TIME 154 μSEC MULTIPLY TIME	14 μSEC ADD TIME 154 μSEC MULTIPLY TIME	14 μSEC ADD TIME 154 μSEC MULTIPLY TIME
INSTRUCTIONS	25	25	25	25
CHECKOUT TEST & SEQUENCE EVENT CONTROL				
TIME WORD RESOLUTION	SELECTABLE 50 m SEC OR 1 SEC	SELECTABLE 50m SEC OR 1 SEC	SELECTABLE 50m SEC OR 1 SEC	SELECTABLE 50m SEC OR 1 SEC
TIME ACCURACY	±0.01% (3σ)	±0.01% (3σ)	±0.01% (3σ)	±0.01% (3σ)
OUTPUT CHANNELS	UP TO 128 OUTPUTS FOR EACH OF CHECKOUT & POST-SEPARATION SEQUENCING	UP TO 128 OUTPUTS FOR EACH OF CHECKOUT & POST-SEPARATION SEQUENCING	UP TO 128 OUTPUTS FOR EACH OF CHECKOUT & POST-SEPARATION SEQUENCING	UP TO 128 OUTPUTS FOR EACH OF CHECKOUT & POST-SEPARATION SEQUENCING

TABLE 1.6-3
CONCEPTUAL DESIGNS – SYSTEM CHARACTERISTICS SUMMARY (Continued)

SYSTEM	CONCEPT I	CONCEPT II	CONCEPT III	CONCEPT V
GUIDANCE & CONTROL (Cont.)				
LANDING RADAR				
BEAM CONFIGURATION	THREE VELOCITY BEAMS, ONE RANGE BEAM AT BEAM GROUP CENTERLINE ALONG ROLL AXIS	THREE VELOCITY BEAMS, ONE RANGE BEAM AT BEAM GROUP CENTERLINE ALONG ROLL AXIS	THREE VELOCITY BEAMS, ONE RANGE BEAM AT BEAM GROUP CENTERLINE ALONG ROLL AXIS	THREE VELOCITY BEAMS, ONE RANGE BEAM AT BEAM GROUP CENTERLINE ALONG ROLL AXIS
TRANSMITTERS	VARACTOR MULTIPLIERS	VARACTOR MULTIPLIERS	VARACTOR MULTIPLIERS	VARACTOR MULTIPLIERS
POWER, FREQUENCY, & MODULATION	VELOCITY: 0.2 W, 10.5 GHz, CW RANGE: 0.1 W, 9.6 GHz, SAWTOOTH FM CW	VELOCITY: 0.2 W, 10.5 GHz, CW RANGE: 0.1 W, 9.6 GHz, SAWTOOTH FM CW	VELOCITY: 0.2 W, 10.5 GHz, CW RANGE: 0.1 W, 9.6 GHz, SAWTOOTH FM CW	VELOCITY: 0.2 W, 10.5 GHz, CW RANGE: 0.1 W, 9.6 GHz, SAWTOOTH FM CW
RECEIVERS	HOMODYNE IN-PHASE & QUADRATURE DETECTION	HOMODYNE IN-PHASE & QUADRATURE DETECTION	HOMODYNE IN-PHASE & QUADRATURE DETECTION	HOMODYNE IN-PHASE & QUADRATURE DETECTION
ANTENNA	INTEGRATED PHASED ARRAY	INTEGRATED PHASED ARRAY	INTEGRATED PHASED ARRAY	INTEGRATED PHASED ARRAY
ACCURACY (3σ TERMINAL PHASE)				
RANGE	±1.4% ±5 FT	±1.4% ±5 FT	±1.4% ±5 FT	±1.4% ±5 FT
VELOCITY	±1.5% ±1.5 FT/SEC	±1.5% ±1.5 FT/SEC	±1.5% ±1.5 FT/SEC	±1.5% ±1.5 FT/SEC
RADAR ALTIMETER	NON-COHERENT PULSE RADAR	NON-COHERENT PULSE RADAR	NON-COHERENT PULSE RADAR	NON-COHERENT PULSE RADAR
AEROSHELL ANTENNA	SINGLE SLOT – 90° ROLL x 160° PITCH	SINGLE SLOT – 90° ROLL x 160° PITCH	SINGLE SLOT – 90° ROLL x 160° PITCH	SINGLE SLOT – 90° ROLL x 160° PITCH
LANDER ANTENNA	CROSSED SLOT – 120° x 120°	CROSSED SLOT – 120° x 120°	CROSSED SLOT – 120° x 120°	CROSSED SLOT – 120° x 120°
TRANSMITTER	TRIODE	TRIODE	TRIODE	TRIODE
FREQUENCY	1.0 GHz	1.0 GHz	1.0 GHz	1.0 GHz
PRF	500 GHz	500 GHz	500 GHz	500 GHz
PULSE WIDTH	5 μSEC	5 μSEC	5 μSEC	5 μSEC
PEAK POWER	500 W	500 W	500 W	500 W
RECEIVER	SUPERHET, RF AMP, LEADING EDGE TRACK	SUPERHET, RF AMP, LEADING EDGE TRACK	SUPERHET, RF AMP, LEADING EDGE TRACK	SUPERHET, RF AMP, LEADING EDGE TRACK
PERFORMANCE				
ALTITUDE	5000 TO 210,000 FT	5000 TO 210,000 FT	5000 TO 210,000 FT	5000 TO 210,000 FT
ACCURACY (3σ)	±0.16% + 630 FT	± 0.16% ± 630 FT	± 0.16% ± 630 FT	± 0.16% ± 630 FT
SEQUENCER				
SURFACE SEQUENCER & TIMER				
TYPE	MAGNETIC CORE MEMORY DEVICE; UPDATEABLE BEFORE SEPARATION	MAGNETIC CORE MEMORY DEVICE; UPDATEABLE BEFORE SEPARATION	MAGNETIC CORE MEMORY DEVICE; UPDATEABLE BEFORE SEPARATION & AFTER LANDING	MAGNETIC CORE MEMORY DEVICE; UPDATEABLE BEFORE SEPARATION & AFTER LANDING
CAPACITY	64 EVENT TIME WORDS & OUTPUTS AVAILABLE	64 EVENT TIME WORDS & OUTPUTS AVAILABLE	128 EVENT TIME WORDS & OUTPUTS AVAILABLE	128 EVENT TIME WORDS & OUTPUTS AVAILABLE
TIME WORD RESOLUTION	1 SEC	1 SEC	1 SEC	1 SEC
TIME ACCURACY	±0.01% (3σ) (±8.9 SEC ACCURACY IN 24.6 HR)	±0.01% (3σ) (±8.9 SEC ACCURACY IN 24.6 HR)	±0.01% (3σ) (±8.9 SEC ACCURACY IN 24.6 HR)	±0.01% (3σ) (±8.9 SEC ACCURACY IN 24.6 HR)
CENTRAL COMPUTER & SEQUENCER				
TYPE	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT TESTING, & POST-SEPARATION TO LANDING SEQUENCING; UPDATEABLE PRIOR TO SEPARATION (SEE G&C SYSTEM SUMMARY)	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT TESTING, & POST-SEPARATION TO LANDING SEQUENCING; UPDATEABLE PRIOR TO SEPARATION (SEE G&C SYSTEM SUMMARY)	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT TESTING, & POST-SEPARATION TO LANDING SEQUENCING; UPDATEABLE PRIOR TO SEPARATION (SEE G&C SYSTEM SUMMARY)	GENERAL PURPOSE COMPUTER FOR G&C COMPUTATIONS, PRE-SEPARATION CHECKOUT TESTING, & POST-SEPARATION TO LANDING SEQUENCING; UPDATEABLE PRIOR TO SEPARATION (SEE G&C SYSTEM SUMMARY)
CAPACITY	128 EVENT TIME WORDS & OUTPUTS AVAILABLE FOR EACH OF C O TEST & SEQUENCE	128 EVENT TIME WORDS & OUTPUTS AVAILABLE FOR EACH OF C O TEST & SEQUENCE	128 EVENT TIME WORDS & OUTPUTS AVAILABLE FOR EACH OF C O TEST & SEQUENCE	128 EVENT TIME WORDS & OUTPUTS AVAILABLE FOR EACH OF C O TEST & SEQUENCE
TIME WORD RESOLUTION	50 mSEC OR 1 SEC	50 mSEC OR 1 SEC	50 mSEC OR 1 SEC	50 mSEC OR 1 SEC
TIME ACCURACY	±0.01% (3σ) (±3.6 SEC ACCURACY IN 10 HR)	±0.01% (3σ) (±3.6 SEC ACCURACY IN 10 HR)	±0.01% (3σ) (±3.6 SEC ACCURACY IN 10 HR)	±0.01% (3σ) (±3.6 SEC ACCURACY IN 10 HR)
THERMAL CONTROL				
FLIGHT PHASE				
TOTAL WEIGHT	47 LB	56.6 LB	60.0 LB	47.9 LB
TEMPERATURE RANGE				
LANDER LIMITS	20° TO 70°F	20° TO 70°F	20° TO 70°F	20° TO 70°F
AEROSHELL ABLATOR	–150°F MIN	–150°F MIN	–150°F MIN	–150°F MIN
PROPELLANTS	40°F MIN	40°F MIN	40°F MIN	40°F MIN
EXTERNAL (INSULATION)				
WEIGHT	19 LB	26.5 LB	29.1 LB	19.4 LB
THICKNESS	0.5 IN.	0.5 IN.	0.5 IN.	0.5 IN.
INTERNAL (HEATERS, COATINGS, INSULATION, ETC.)				
WEIGHT	28 LB	30.1 LB	30.9 LB	28.5 LB

TABLE 1.6-3
CONCEPTUAL DESIGNS – SYSTEM CHARACTERISTICS SUMMARY (Continued)

SYSTEM	CONCEPT I	CONCEPT II	CONCEPT III	CONCEPT V
THERMAL CONTROL (Cont.)				
LANDED PHASE				
TOTAL WEIGHT	37.8 LB	24.8 LB	49.6 LB	42.7 LB
OUTER EQUIPMENT COMPARTMENT				
TEMPERATURE RANGE	0° TO 100°F	0° TO 100°F	0° TO 100°F	0° TO 100°F
INSULATION DENSITY	4 LB/FT^3	4 LB/FT^3	4 LB/FT^3	4 LB/FT^3
INSULATION THICKNESS	3.0 IN.	3.0 IN.	3.0 IN.	3.0 IN.
HEATER TYPE	ISOTOPE (7.5 W)	ELECTRICAL (219 WH)	ISOTOPE (12.5 W)	ISOTOPE (10 W)
HEAT SINK	BERYLLIUM (1.4 LB)	NONE	NONE	NONE
INNER BATTERY COMPARTMENT				
TEMPERATURE RANGE	50° TO 125°F	50° TO 125°F	50° TO 125°F	50° TO 125°F
INSULATION DENSITY	4.0 LB/FT^3	4.0 LB/FT^3	4.0 LB/FT^3	4.0 LB/FT^3
INSULATION THICKNESS	0.4 IN.	0.4 IN.	0.4 IN.	0.4 IN.
HEATER TYPE	ISOTOPE (22 W)	ELECTRIC (1763 WH)	ISOTOPE (51W) & ELECTRIC (680 WH–ONE DAY)	ISOTOPE (25.5 W)
HEAT SINK	BERYLLIUM (7.4 LB)	NONE	HYDROCARBON WAX (7.6 LB)	BERYLLIUM (8.1 LB)
PROPULSION				
DEORBIT				
NUMBER OF MOTORS	1	1	1	1
SYSTEM WEIGHT	62.1 LB	73 LB	80.7 LB	66.2 LB
TOTAL IMPULSE	15,000 LB-SEC	17,620 LB-SEC	19,600 LB-SEC	16,060 LB-SEC
THRUST	1190 LB	1325 LB	1420 LB	1244 LB
PROPELLANT	PBAP/AL	PBAP/AL	PBAP/AL	PBAP/AL
VACUUM I_{sp}	288 SEC	288 SEC	288 SEC	288 SEC
MOTOR LENGTH	18.1 IN.	19.2 IN.	19.9 IN.	18.5 IN.
ATTITUDE CONTROL				
NUMBER OF THRUSTERS	8 (4 PITCH/YAW, 4 ROLL)	8 (4 PITCH/YAW, 4 ROLL)	8 (4 PITCH/YAW, 4 ROLL)	8 (4 PITCH/YAW, 4 ROLL)
SYSTEM WEIGHT	30.9 LB	31.8 LB	32.6 LB	31.2 LB
TOTAL IMPULSE	310 LB-SEC	320 LB-SEC	330 LB-SEC	315 LB-SEC
THRUST	7.4 LB P/Y, 0.5 LB R	7.6 LB P/Y, 0.5 LB R	7.7 LB P/Y, 0.5 LB R	7.5 LB P/Y, 0.5 LB R
PROPELLANT	N_2	N_2	N_2	N_2
VACUUM I_{sp}	72.5 SEC	72.5 SEC	72.5 SEC	72.5 SEC
TERMINAL				
NUMBER OF ENGINES	3	3	3	3
SYSTEM WEIGHT *	198 LB	214 LB	229 LB	208 LB
TOTAL IMPULSE	16,700 LB-SEC	19,000 LB-SEC	21,100 LB-SEC	18,200 LB-SEC
THRUST/ENGINE	231 TO 61 LB	325 TO 68 LB	354 TO 75 LB	252 TO 66 LB
PROPELLANT	N_2H_4	N_2H_4	N_2H_4	N_2H_4
NOZZLE EXPANSION RATIO	30	30	30	30
VACUUM I_{sp}	233.5 SEC	233.5 SEC	233.5 SEC	233.5 SEC
AUXILIARY AERODYNAMIC DECELERATOR				
TYPE	MODIFIED RINGSAIL	MODIFIED RINGSAIL	MODIFIED RINGSAIL	MODIFIED RINGSAIL
WEIGHT AND SIZE:				
NOMINAL DIAMETER, D_o	30.1 FT	31.7 FT	34.0 FT	31.5 FT
SYSTEM WEIGHT	34.3 LB	40.1 LB	45.5 LB	39.0 LB
DECELERATOR LOAD	779 LB	871 LB	975 LB	853 LB
BALLISTIC PARAMETER	.0486 SLUGS FT^2	.0487 SLUGS FT^2	.0474 SLUGS FT^2	.0486 SLUGS FT^2
AEROSHELL				
STRUCTURE	TITANIUM SINGLE-FACED LONGITUDINALLY CORRUGATED CONIC SHELL & A REINFORCED PLASTIC SANDWICH NOSE ASSEMBLY	TITANIUM SINGLE-FACED LONGITUDINALLY CORRUGATED CONIC SHELL & A REINFORCED PLASTIC SANDWICH NOSE ASSEMBLY	TITANIUM SINGLE-FACED LONGITUDINALLY CORRUGATED CONIC SHELL & A REINFORCED PLASTIC SANDWICH NOSE ASSEMBLY	TITANIUM SINGLE-FACED LONGITUDINALLY CORRUGATED CONIC SHELL & A REINFORCED PLASTIC SANDWICH NOSE ASSEMBLY
CONFIGURATION	120° SPHERE-CONE	120° SPHERE-CONE	120° SPHERE-CONE	120° SPHERE-CONE
WEIGHT	52.9 LB	91.2 LB	102.2 LB	54.1 LB
BASE DIAMETER	8.83 FT	10.9 FT	11.48 FT	8.83 FT
NOSE RADIUS – SPHERICAL	2.21 FT	2.73 FT	2.87 FT	2.21 FT
ENTRY HEAT PROTECTION				
NOSE CAP, NON-ABLATIVE	HARDENED COMPACTED FIBERS (HCF) OF ALUMINO-SILICATE; 25 LB/FT^3	HARDENED COMPACTED FIBERS (HCF) OF ALUMINO-SILICATE; 25 LB/FT^3	HARDENED COMPACTED FIBERS (HCF) OF ALUMINO-SILICATE; 25 LB/FT^3	HARDENED COMPACTED FIBERS (HCF) OF ALUMINO-SILICATE; 25 LB/FT^3
CONICAL AEROSHELL ABLATOR	MCDONNELL S–20T; 20 LB/FT^3 FOAMED SILICONE IN HONEYCOMB	MCDONNELL S–20T; 20 LB/FT^3 FOAMED SILICONE IN HONEYCOMB	MCDONNELL S–20T; 20 LB/FT^3 FOAMED SILICONE IN HONEYCOMB	MCDONNELL S–20T; 20 LB/FT^3 FOAMED SILICONE IN HONEYCOMB
AFT THERMAL CURTAIN	COATED SILICA CLOTH; 0.1 LB/FT^2 ; 11.0 LB	COATED SILICA CLOTH; 0.1 LB/FT^2 ;13.4 LB	COATED SILICA CLOTH; 0.1 LB/FT^2 ; 14.6 LB	COATED SILICA CLOTH; 0.1 LB/FT^2 ; 11.6 LB
ADHESIVES FOR HCF & S–20T	HT–424	HT–424	HT–424	HT–424
WEIGHT (W/O AFT CURTAIN)	60.1 LB	82.8 LB	90.9 LB	60.9 LB
LANDING SYSTEM				
UNI-DISC				
FOOTPAD DIAMETER	55.7 IN.	63.7 IN.	66.3 IN.	56.8 IN.
IMPACT ATTENUATOR HEIGHT	10 IN.	10 IN.	10 IN.	10 IN.
IMPACT ATTENUATOR DIAMETER	38.4 IN.	43.7 IN.	45.4 IN.	39.1 IN.
TOTAL WEIGHT	105.6 LB	128.4 LB	138.6 LB	111.1 LB
CANISTER & ADAPTER				
CANISTER				
TOTAL WEIGHT	129.6 LB	174.0 LB	190.0 LB	132.1 LB
MAXIMUM DIAMETER	109.5 IN.	138.3 IN.	145.3 IN.	109.5 IN.
MAXIMUM HEIGHT	82.6 IN.	87.2 IN.	90.0 IN.	84.85 IN.
ADAPTER				
TOTAL WEIGHT	15.2 LB	16.2 LB	17.3 LB	15.8 LB

*WEIGHTS BASED ON USE OF EXISTING HARDWARE e.g., W. KIDDE ENGINE and cost efficiency. If the development of sterilizable solid propellant systems does not proceed promptly, a liquid propulsion system using components common with the terminal propulsion system can be incorporated. Although this would reduce the total cost of the propulsion elements, the weight penalties incurred would increase the size and cost of other systems and result in a net cost increase for the total capsule.

The probability of mission success for each of these concepts depends on the reliability of the equipment to perform in the design environments and the capsule's invulnerability to off-nominal conditions which may be encountered. These conceptual designs do not include redundant components. This design approach was selected to facilitate concept comparison and to ultimately assure the most efficient approach to provide a high reliability capsule. In using a similar approach for the VOYAGER Capsule Phase B design (see NASA/CR 89672), it was determined that a gross weight increase of 73 lbs for redundant components increased reliability for performing all functions to .71. A similar reliability could be obtained for these four conceptual designs by adding the redundant components listed in Table 1.6-4. The capsule weight would be increased by these additions and by the weight required for the delivery of these additional components. In Concepts II and III this would require a small increase in capsule diameter. In Concepts I and V the 8.83 ft diameter is sufficient to accomodate this additional weight.

The sensitivity of mission success probability to the environment is different for each concept. Concepts I and V are designed to a less-than-nominally-severe environment. If any of the extremes of the nominal environment is encountered, probability of mission success is reduced. However, since many capsule systems are designed for a combination of worst conditions, the violation of any single limit may be tolerable. Concepts III and V utilize solar cells for power, and they are susceptible to environmental degradation. Table 1.6-5 presents the environment sensitivities for each concept.

Configuration and weight distribution of Concept V, a 1300 lb capsule with extended lifetime, are shown in Figure 1.6-1 and Table 1.6-6 respectively. In this configuration, the aeroshell diameter is 106 inches, and the weight at entry is 1050 lb; this results in a ballistic parameter of .35 slug/ft^2 and

TABLE 1.6–4
REDUNDANCY CANDIDATES

	CONCEPT I		CONCEPT II		CONCEPT III		CONCEPT V	
	COMPONENT WEIGHT	EFFECT ON CAPSULE WEIGHT	COMPONENT WEIGHT	EFFECT ON CAPSULE WEIGHT	COMPONENT WEIGHT	EFFECT ON CAPSULE WEIGHT	COMPONENT WEIGHT	EFFECT ON CAPSULE WEIGHT
SURFACE PAYLOAD								
REDUNDANT TWT	–		–		4.0		4.0	
REDUNDANT TWT DRIVERS	–		–		1.0		1.0	
REDUNDANT A/D CONVERTERS	–		–		.5		.5	
REDUNDANT POWER INTERCONNECTIONS	1.5		1.5		1.5		1.5	
EQUIPMENT DEPLOY & RELEASE PYRO DEVICES (4)	1.0		1.0		1.0		1.0	
PAYLOAD WEIGHT INCREMENT	2.5		2.5		8.0		8.0	
LANDER SYSTEMS								
DC-DC CONVERTER REGULATOR	–		4.0		–		–	
MODIFY RANGE CHANNEL TO TIME SHARE AS BOTH RANGE & VELOCITY CHANNEL	1.5		1.5		1.5		1.5	
ACTIVE REDUNDANT RECEIVERS & TRACKERS IN THE RADAR ALTIMETER	4.3		4.3		4.3		4.3	
RADAR ALTIMETER TRANSMITTER TUBE	1.44		1.44		1.44		1.44	
REDUNDANT POWER INTERCONNECTIONS	1.25		1.25		1.25		1.25	
LANDER SYSTEMS WEIGHT INCREMENT	8.49		12.49		8.49		8.49	
LANDER WEIGHT INCREMENT	10.99	21.10	14.99	32.83	16.49	36.11	16.49	31.66
ENTRY SYSTEMS								
AEROSHELL RELEASE CARTRIDGE TO EXPLOSIVE BOLT ASSEMBLY	1.48		1.48		1.48		1.48	
PARACHUTE CATAPULT INITIATOR	.25		.25		.25		.25	
PARACHUTE RELEASE CARTRIDGE TO EXPLOSIVE BOLT ASSEMBLY	1.48		1.48		1.48		1.48	
ENTRY SYSTEMS WEIGHT INCREMENT	3.21	3.85	3.21	3.92	3.21	3.92	3.21	3.85
ENTRY WEIGHT INCREMENT		24.95		36.75		40.03		35.51
DEORBIT MOTOR								
MOTOR RELEASE CARTRIDGE TO EXPLOSIVE BOLT ASSEMBLY	1.48	1.78	1.48	1.81	1.48	1.81	1.48	1.78
DEORBIT WEIGHT INCREMENT		26.73		38.56		41.84		37.29
CANISTER/ADAPTER EQUIPMENT								
CAPSULE/ADAPTER SEPARATION CARTRIDGE TO EXPLOSIVE BOLT ASSEMBLY	2.96		2.96		2.96		2.96	
SMDC FORWARD CANISTER RELEASE	7.05		9.00		9.75		7.05	
CANISTER/ADAPTER WEIGHT INCREMENT	10.01	11.05	11.96	13.75	12.71	14.62	10.01	11.05
TOTAL CAPSULE WEIGHT INCREMENT		37.78		52.31		56.46		48.34

TABLE 1.6–5
CAPSULE SENSITIVITY TO ENVIRONMENT

ENVIRONMENT ENCOUNTERED	DESIGN CONCEPT I	DESIGN CONCEPT II	DESIGN CONCEPT III	DESIGN CONCEPT V
1) SURFACE TEMPERATURE 150°K	REDUCED MISSION TIME DUE TO EARLY BATTERY ENERGY DEPLETION. THERMAL CONTROL SYSTEM MAY BE MARGINAL.	NONE	ONE DIURNAL CYCLE OF OPERATION, IF 150°K IS ENCOUNTERED, CAN BE ACHIEVED WITHOUT COMPROMISING THE BASIC MISSION PROFILE. MISSION EXTENSION BEYOND ONE DAY OF 150°K MAY BE POSSIBLE IF THE MISSION PROFILE IS ALTERED.	MISSION TIME WILL BE REDUCED DUE TO EARLY DEPLETION OF BATTERY ENERGY. QUANTITY OF DATA RECEIVED MAY BE REDUCED. THERMAL CONTROL SYSTEM MAY BE MARGINAL.
172°K TO 322°K (CYCLIC DAY)	NONE	NONE	NONE	NONE
2) SURFACE SLOPES 20°	NONE	NONE	NONE	NONE
34°	COMMUNICATIONS TO ORBITER MAY BE RESTRICTED DUE TO SHORTER VIEW TIMES.	NONE	NONE	COMMUNICATIONS TO ORBITER MAY BE RESTRICTED DUE TO SHORTER VIEW TIMES. SOLAR PANEL PERFORMANCE MAY BE REDUCED.
3) SURFACE WINDS 118 FT/SEC	NONE	NONE	NONE	NONE
220 FT/SEC	TERMINAL PROPULSION MAY BE MARGINAL IN PROVIDING THE REQUIRED THRUST LEVEL. MAINTAINING RADAR LOCK MAY BE MARGINAL DUE TO EXCESSIVE CAPSULE ATTITUDES.			TERMINAL PROPULSION MAY BE MARGINAL IN PROVIDING THE REQUIRED THRUST LEVEL. MAINTAINING RADAR LOCK MAY BE MARGINAL DUE TO EXCESSIVE CAPSULE ATTITUDES.
4) SCALE HEIGHT 8 KM TO 14 KM	NONE	NONE	NONE	NONE
5 KM TO 14 KM	FOR SCALE HEIGHT LESS THAN 8 KM THE LOADS EXPERIENCED BY PARACHUTE MAY EXCEED DESIGN LIMITS.			FOR SCALE HEIGHT LESS THAN 8 KM THE LOADS EXPERIENCED BY PARACHUTE MAY EXCEED DESIGN LIMITS.
5) ATMOSPHERIC PRESSURE 5 MB TO 20 MB	NONE	NONE	NONE	NONE
6) CLOUDS	NONE	NONE	SOLAR PANEL PERFORMANCE MAY BE MARGINAL OR DEGRADED DUE TO ATMOSPHERIC ATTENUATION. RECEIVED DATA MAY BE REDUCED.	SOLAR PANEL PERFORMANCE MAY BE MARGINAL OR DEGRADED DUE TO ATMOSPHERIC ATTENUATION. RECEIVED DATA MAY BE REDUCED.
7) DUST	NONE	NONE	DEPOSITS ON SOLAR PANELS MAY CAUSE DEGRADED PERFORMANCE OF SOLAR CELLS. RECEIVED DATA MAY BE REDUCED.	DEPOSITS ON SOLAR PANELS MAY CAUSE DEGRADED PERFORMANCE OF SOLAR CELLS. RECEIVED DATA MAY BE REDUCED.

FLIGHT CAPSULE
GENERAL ARRANGEMENT
CONCEPT V

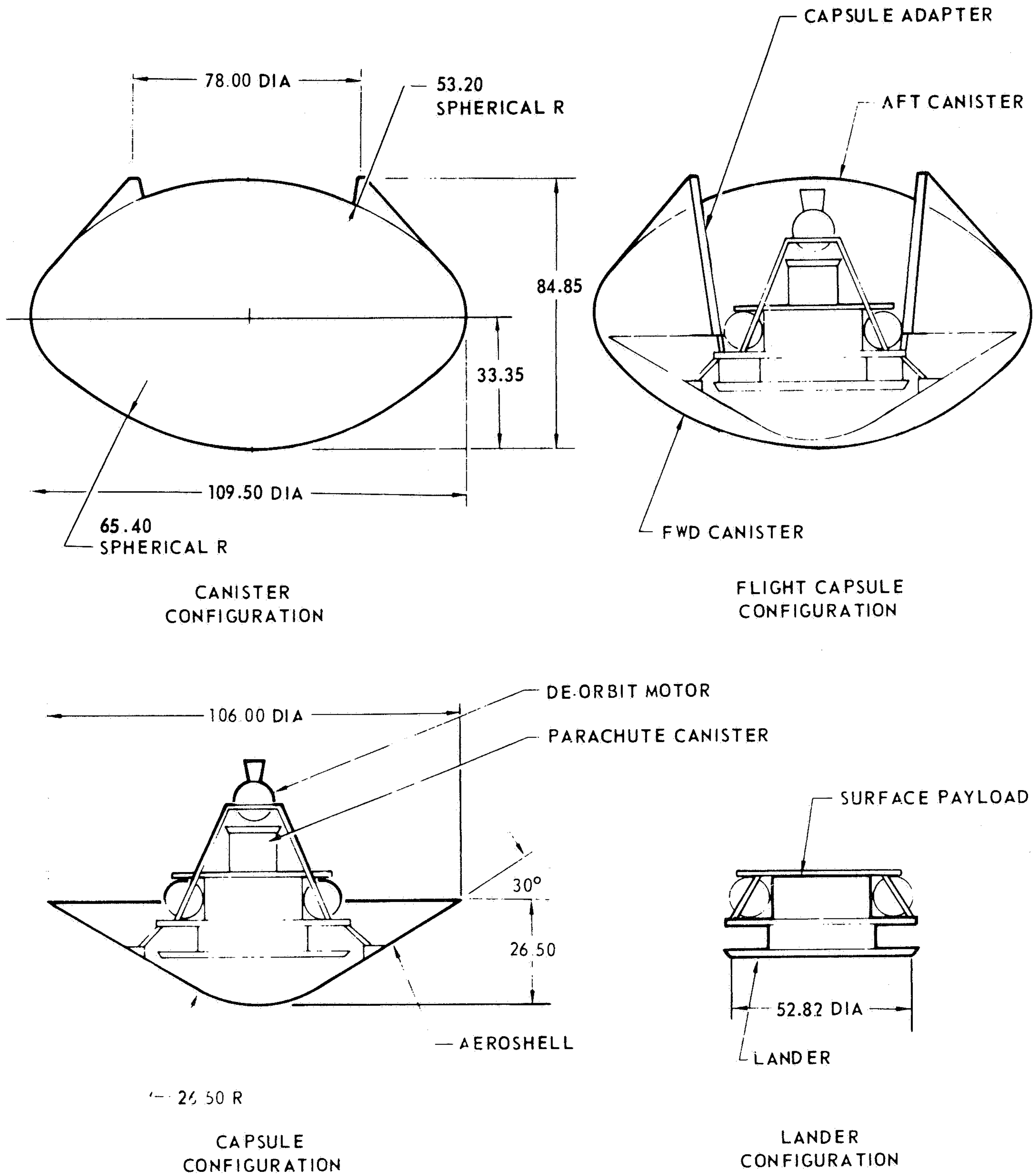


FIGURE 1.6–1

TABLE 1.6-6
CONCEPT V WEIGHT SUMMARY

	PARAMETRIC	POINT DESIGN
STERILIZATION CANISTER	132	132
ADAPTER	16	16
DEORBIT PROPULSION	((76))	((74))
PROPELLANT	53	56
INERT MOTOR	15	10
MISCELLANEOUS	8	8
AEROSHELL	((115))	((115))
STRUCTURE	54	54
HEAT SHIELD	61	61
STRUCTURE	((165))	((160))
INTERNAL	(19)	(19)
ADAPTER SEPARATION PROVISIONS	9	9
LANDER SEPARATION PROVISIONS	10	10
LANDING SYSTEM	111	111
SURFACE PAYLOAD	34	30
THERMAL CONTROL	((98))	((102))
CANISTER	19	19
AEROSHELL	3	3
THERMAL CURTAIN	12	12
LANDER	25	25
SURFACE PAYLOAD	39	43
FLIGHT CONTROL SYSTEM	((158))	((156))
ATTITUDE CONTROL SYSTEM	(42)	(36)
PROPELLANT	5	4
HARDWARE AND MOUNTING PROVISIONS	32	27
WIRING	5	5
ELECTRONICS	(116)	(120)
RADAR	55	57
GUIDANCE SYSTEM	37	37
WIRING AND MOUNTING PROVISIONS	24	26
PARACHUTE	((39))	((39))
CLOTH AND LINES	23	23
DEPLOYMENT, ATTACH, AND MISC.	16	16
TERMINAL PROPULSION	((193))	((214))
PROPELLANT	85	84
GAS(PRESSURANT)	1	1
HARDWARE AND MOUNTING PROVISIONS	101	123 (1)
WIRING	6	6
POWER & SEQUENCER	((137))	((141))
EQUIPMENT	114	109
WIRING AND MOUNTING PROVISIONS	23	32
TELECOMMUNICATIONS (INCL. INSTRUMENTATION)	((117))	((97))
EQUIPMENT	92	73 (2)
WIRING AND MOUNTING PROVISIONS	25	24
SCIENCE	((67))	((69))
EQUIPMENT	53	53
WIRING AND MOUNTING PROVISIONS	14	16
TOTAL CAPSULE WEIGHT	1313	1315
SEPARATED CAPSULE WEIGHT	1120	1138
ENTRY WEIGHT	1050	1069
DECELERATOR LOAD	853	879
LANDED WEIGHT	739	758
SURFACE PAYLOAD WEIGHT	226	244

NOTES: (1) USED EXISTING HARDWARE
(2) EQUIPMENT INTEGRATED – SPECIFIC DATA REQUIREMENTS RE-EVALUATED.

a Mach 1.73 parachute deployment at the nominal deployment altitude of 23 000 feet. The Concept V interior arrangement presented in Figure 1.6-2 shows the location of equipment in the aeroshell and canister. Location of the lander equipment is shown in Figure 1.6-3.

FLIGHT CAPSULE INTERIOR ARRANGEMENT

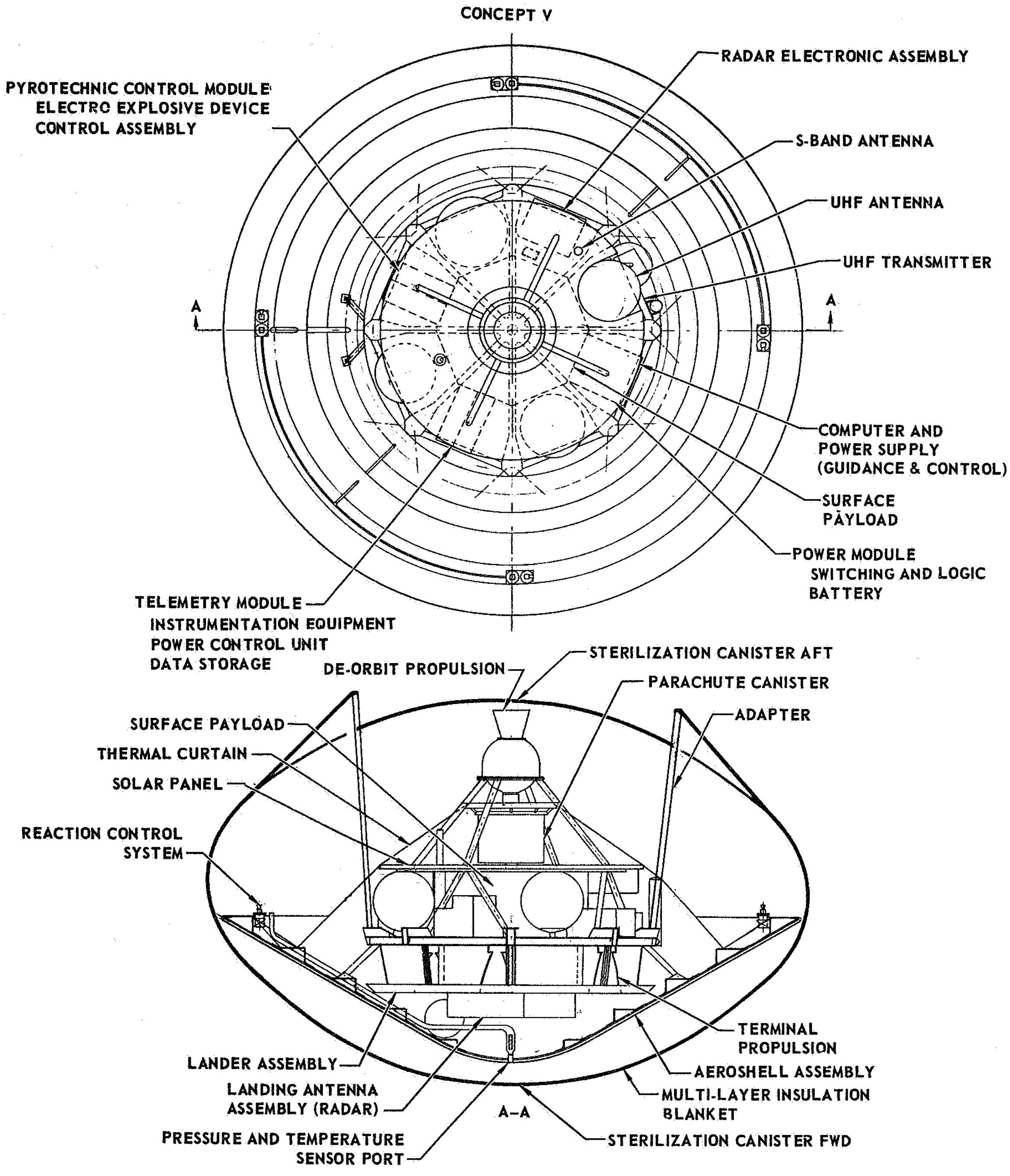


FIGURE 1.6–2

LANDER GENERAL ARRANGEMENT

CONCEPT V

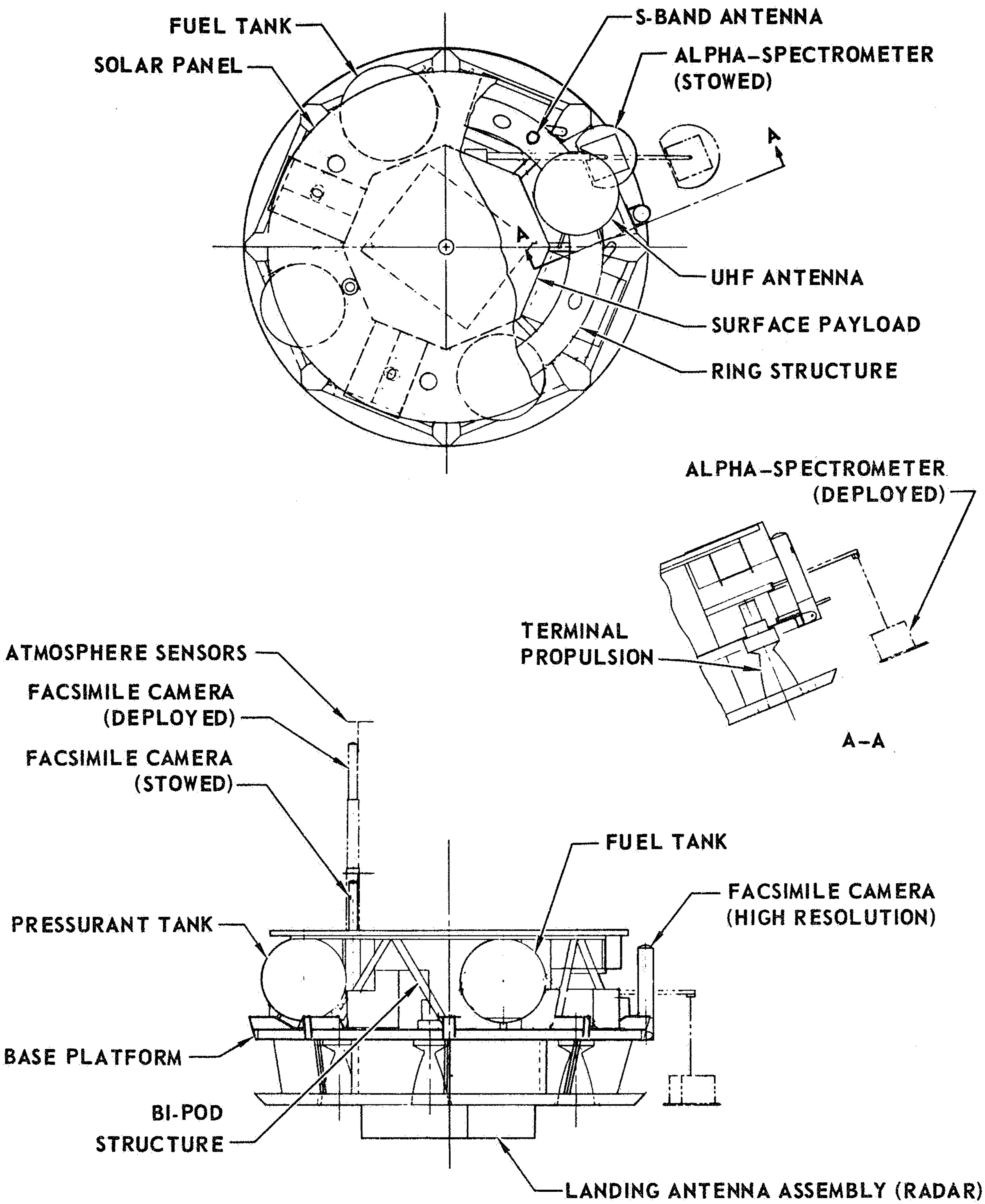


FIGURE 1.6-3

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

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